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USE OF STS SUBSYSTEMS AND COMPONENTS FOR MMSE

FINAL TECHNICAL REPORT

DECEMBER 1975

Volume 2

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Space Division
Rockwell International

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FOREWORD

Multiuse Mission Support Equipment (MMSE) is that flight/ground equipment for the Shuttle era which is used in conjunction with more than one mission payload. It is expected to be used repeatedly with appropriate refurbishment between uses.

This study provides NASA with data verifying STS subsystems applicability to MMSE, along with the cost savings potential and the programmatic data needed for further program planning decisions.

Some 70 MMSE requirements were found to be potentially satisfied by STS equipment, and six items of particular interest were chosen for special emphasis in this study. All were found to be feasible and beneficial to NASA. Program cost savings through their use is estimated to be substantial; approximately $200 million can be saved over 10 years by use of the STS subsystems and components to fulfill presently identified MMSE requirements. This savings becomes more than $400 million by implementing the STS multiple launch capability for Thor-Delta payloads with utilization of MMSE payload spin-up mechanisms.

Considering the potential savings involved, it is strongly recommended that the study be continued to identify additional MMSE requirements and hardware. Detailed definition studies are recommended for FY 76 in support of needed procurements in FY 77.

The work described in this final report was performed under a $75,000 contract, NAS9-14598, for NASA Johnson Space Center. The NASA Technical Monitor (COR) was L. J. Nado and the Rockwell Study Manager was J. O. Matzenauer. Any questions concerning the material presented can be addressed to either of these individuals.

The contract required mid-term and final briefings and reports. This technical report volume contains the detailed technical data and results in NASA reporting format. The final briefing presentation is identified as SD 75-SA-0182 and the final Executive Summary Report as SD 75-SA-0181, Volume 1.

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TABLES

1. INTRODUCTION

Multiuse Mission Support Equipment (MMSE) is that ground or flight payload support equipment for the Shuttle era which is used in conjunction with more than one mission payload, and is likely to be utilized repeatedly with appropriate refurbishment between uses. Thus, the STS equipment should be a fruitful source of equipment/subsystems/components in view of its design for multiple reuse, its concurrency permitting simplified procurement, known interfaces, and the indirect benefits to STS which could result from a broader cost and utilization base. The effort is in line with a general trend to utilize standardized equipment concepts for low cost development and operation.

The overall objective of the study was to generate data to provide NASA with initial verification of STS subsystems applicability to MMSE along with the cost savings potential and the programmatic data on key items needed for further program planning decisions. This is an important step, not only from the cost-savings standpoint but also to permit timely planning for procurement of needed items with maximum benefit to both STS and MMSE programs.

The approach was to utilize the recent MMSE study, Contract NAS8-30847, as an initial source of MMSE requirements, supplemented by the contractor's own payload accommodation findings evolving from studies of major Shuttle payloads such as EOS, LST, Spacelab, and Tug/IUS, and the specific interfacing of these payloads with the Shuttle Orbiter. The problem was to gather and correlate from diverse sources the necessary STS ground and flight subsystems/components data for use in satisfying the MMSE requirements. It is notable that the Shuttle Orbiter Program, is further advanced in definition than the other major elements of the STS (SRB, ET, Spacelab, Tug/IUS, ground systems). Therefore, a majority of STS subsystems applicable to MMSE are at this time defined from the Shuttle Orbiter Program. Much of the data needed is not published but is found in internal documents and from personal discussions with knowledgeable individuals. In this way, up-to-the-minute data on Shuttle actual or anticipated developments was obtained and utilized. Perhaps more important, the reality of apparent MMSE requirements could be realistically challenged along with some of the proposed implementation concepts.

The overall simplified study logic can be seen in summary form in Figure 1-1. A very large number of requirements were developed and defined from two general sources and in Task 1 were put into the Shuttle Work Breakdown Structure (WBS) form for ease in later matching STS equipment. These two sources were the previous MSFC-sponsored MMSE study and the contractor's own accumulated experience in studies of many spacecraft, payloads, and interface problems. The requirements fell into two basic categories, ground support equipment (GSE) and flight or airborne support equipment (ASE). The input data varied considerably between these two categories in terms of prior study emphasis and current definition status.

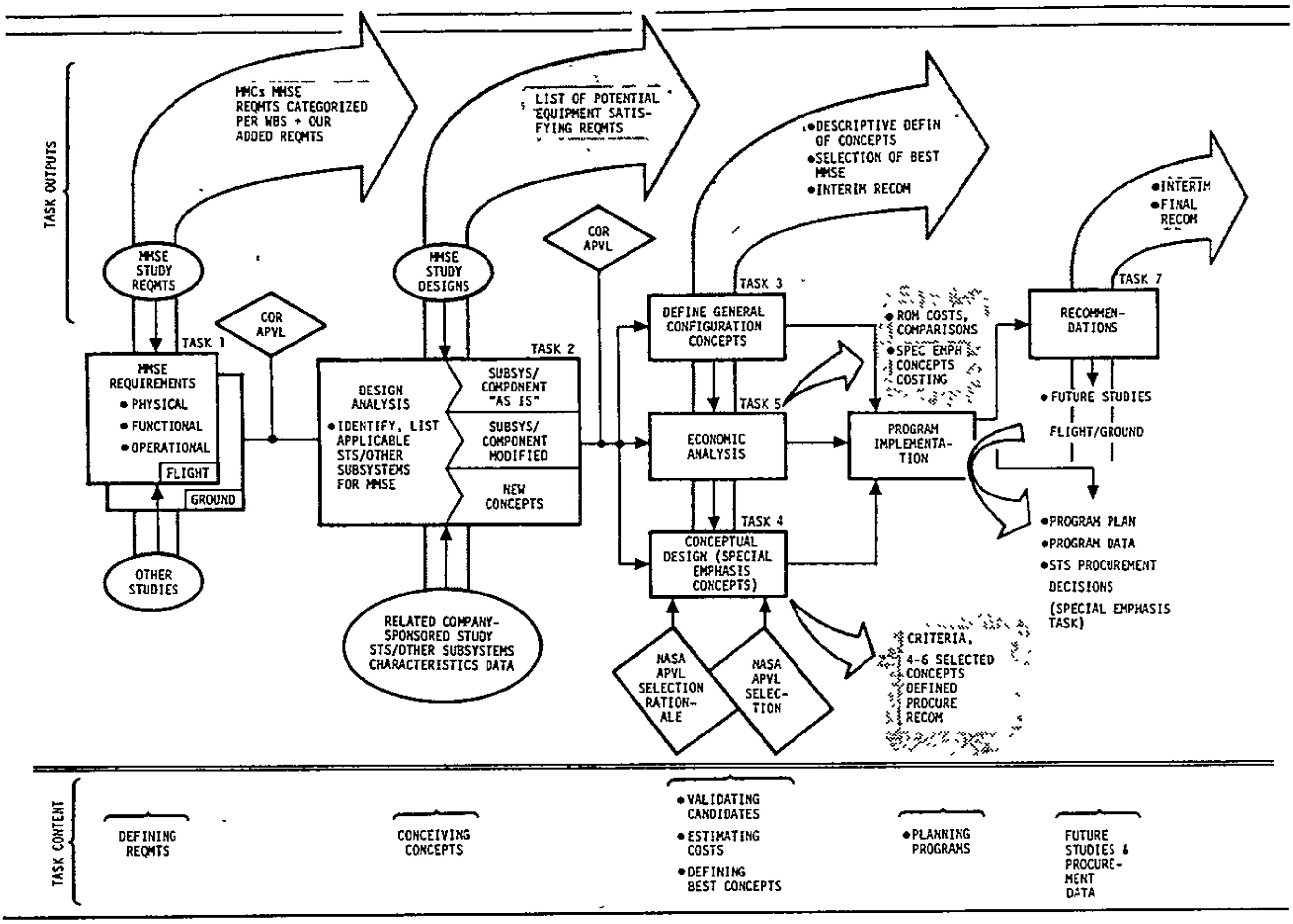


Figure 1-1. Overall Logic

In Task 2, Design Analysis, concepts were identified utilizing STS equipment to satisfy the Task 1 requirements. Thus, the STS equipment for potentially satisfying the MMSE requirements was identified in Task 2 for the previously identified requirements and also for the newly identified requirements. STS equipment (major components/subsystems) was utilized either literally (as-is) or with relatively minor modifications compared to new equipment development. The output of a related company-sponsored study of Shuttle equipment candidates suitable for other applications was of material help in this task in that the time required to research a large amount of data was not necessary, the company-sponsored effort had already done this and presented the data in easy-to-assimilate form.

In Task 3, a more careful look was made at each of the many candidate concepts and particularly at those requirements for which more than one STS equipment concept was possible. Simple economic analyses and engineering judgment was applied in choosing the best of alternate ways to satisfy a

given requirement. Descriptive data sheets were prepared for each "survivor", a total of 70 items. These best concepts were then an input to Task 4 as candidate special emphasis items.

Two special emphasis items were chosen early in the study because of their obvious appeal as STS/MMSE. These are the Orbiter star tracker for payloads and the simplified payload version of the Orbiter Multiplexer-Demultiplexer (MDM). Four more were analyzed in the second half of the study. These are: Multi-discipline Auxiliary Payload Power System (initially conceived in a company-sponsored effort), Payload Spin-up Mechanisms, Payload-to-Orbiter Electric Cables, and Payload-to-Orbiter Fluid Lines. Preliminary concepts, descriptive, and programmatic data for these six special emphasis task items are presented in this report.

From these studies, technical feasibility and programmatic conclusions were to be drawn for the special emphasis items and detailed preliminary study plans were to be provided in the case of four subjects which were too complex in nature for consideration as special emphasis tasks.

The sequence of tasks is pictured in Figure 1-2.

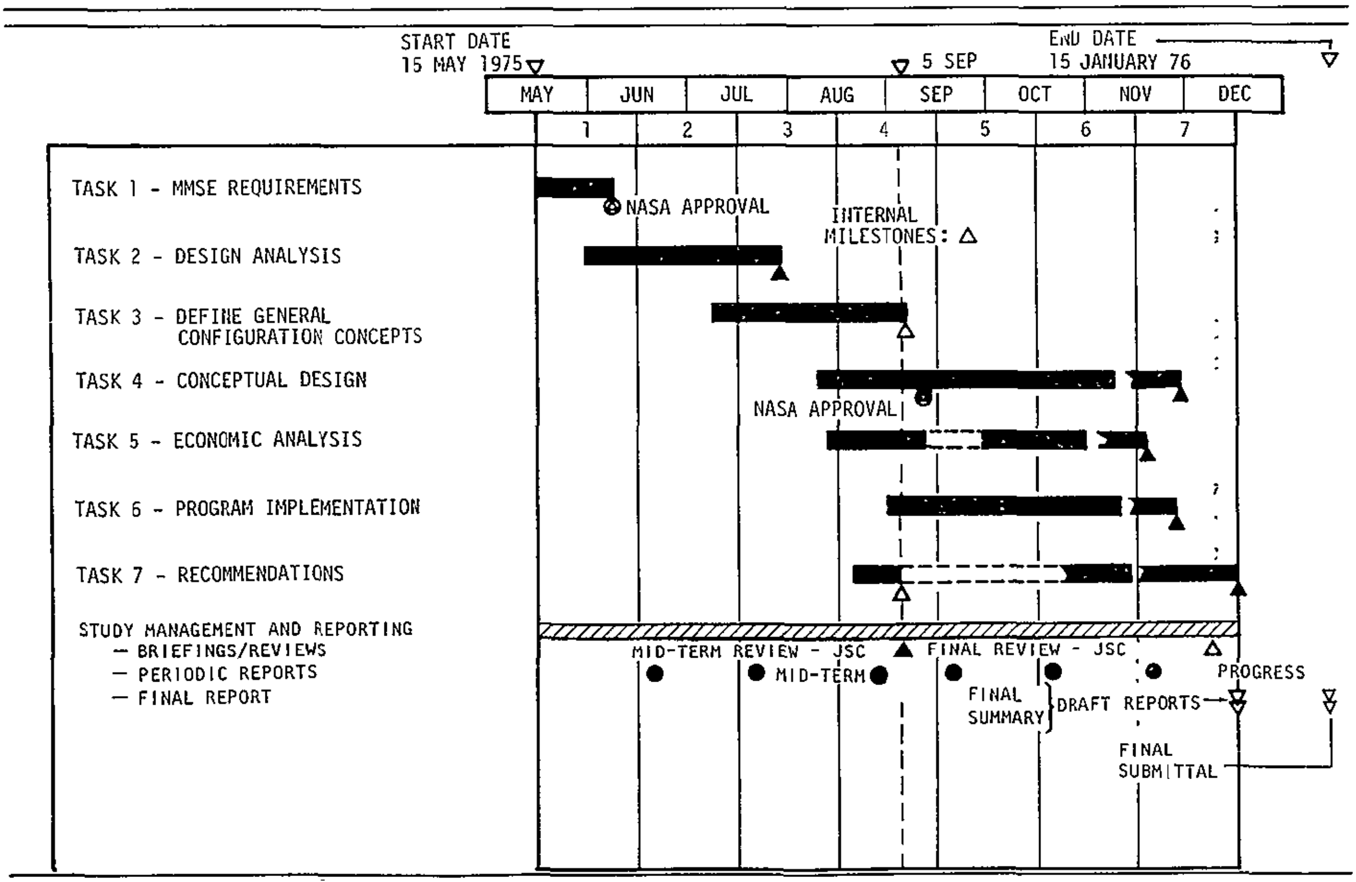


Figure 1-2. Study Task Schedule and Milestones

2. RESULTS

The following results of the study are presented by task and follow in general the sequence as the work was performed.

TASK 1 MMSE REQUIREMENTS

The study started with a detailed review of the MSFC and KSC MMSE study documentation. The final report of each was reviewed and correlated with the interim progress reports to provide a more complete understanding of the nature, validity, and concept for satisfying the requirement and the interrelationship between the many--275 (180 GSE and 95 ASE) initial requirements and the considerably fewer--158 (58 GSE and 100 ASE) final requirements as identified in these studies.

To complete the lists of ASE and GSE requirements, the STS and other programs were reviewed for new MMSE requirements. This review identified seven new requirements for MMSE which are either documented (Auxiliary Power System and Satellite Spin-Up Mechanism) or are readily recognizable as MMSE. They are listed in Table 2-1 and discussed in the following paragraphs.

Table 2-1. New "Hard" MMSE Requirements

1. Auxiliary integrated power system
2. EVA tool kit
3. Satellite spin-up mechanism
4. Orbiter/Spacelab system simulator
5. Payload multiplexer-demultiplexer
6. RMS end effectors
7. Payload integrated pointing system

In the previous MMSE study the auxiliary power system was identified as payload unique as space processing was the only discipline shown by the 1974 Shuttle System Payload Description Study Documents (SSPD) requiring power beyond the current Shuttle capability. The 1975 revision of the SSPD identified two additional payloads with high power requirements. Figure 2-1 presents the latest payload power requirements and the number of missions currently planned (1975 SSPD) and compares these requirements to the Orbiter/Spacelab capability. As noted, the 7 kw available to the payloads from the payload-dedicated fuel cell is not totally available to the experiments. All of the identified payloads are of the sortie class thus requiring some form of the Spacelab (manned pressurized volume and/or pallet with an igloo). The power requirements for the Spacelab systems varies from 1.8 to 3 kw* continuous depending on its

*Late December ESA discussions indicate these numbers should be higher; 2.0 to 3.5 kw

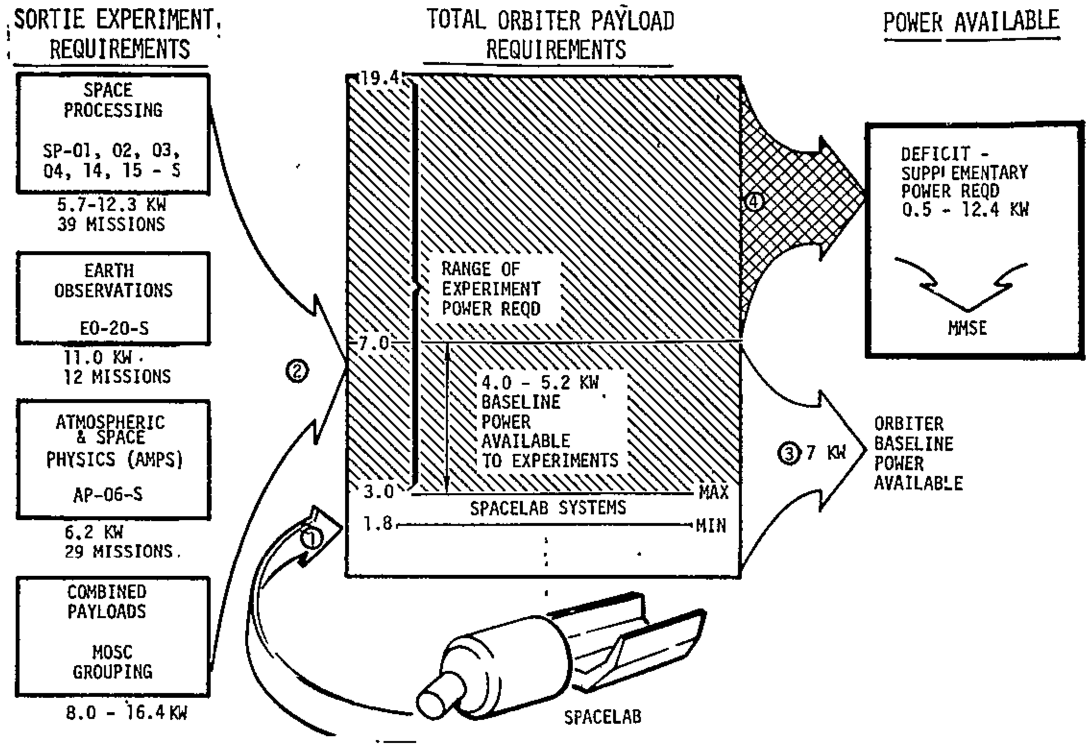


Figure 2-1. Auxiliary Integrated Power System

configuration. Therefore, as shown on Figure 2-1, only 4.0 to 5.2 kw is actually available to the experiments. While the individual payloads (space processing, earth observation, and atmospheric and space physics) require power considerably above the 5.2 maximum available from the Orbiter, the key driver for an auxiliary power system may be the grouped payloads. The Manned Orbital Systems Concept study identifies many grouped payloads. One in particular shows a maximum power requirement of 16.4 kw. While this load is not continuous, it is expected to be of sufficient duration to greatly influence the design of an auxiliary power system. Even though this is a single example, it is indicative of the high power requirements resulting from the grouping of payloads. The potential application of commercial equipment with its high power requirements (approximately two to three times that for a corresponding piece of space-type hardware) further solidifies the need for a mission kit(s) to augment the basic Orbiter power generation capability.

The need for a satellite spin-up mechanism came about because several automated payloads require spin stabilization to accomplish their mission objectives. From a payload standpoint this can best be accomplished by providing the necessary rotational rate before the payload is deployed from the Orbiter. Two of the payloads are currently only planned for the Orbital Flight Test (OFT) program; however, one payload (AP-06-S) in the Shuttle operational phase and the expected future payload requirements justify the spin mechanism as MMSE.

Most payload disciplines desire the capability to simulate (by computer and/or mechanical means) the Orbiter/Spacelab systems associated with the payloads (e.g., payload specialist station and general purpose computer) at their facilities for development of their payloads and for final checkout prior to shipment to the payload integrator. The availability of a piece(s) of MMSE of this nature, which could be used by the various payload disciplines, would provide confidence of Shuttle/payload functional compatibility after installation of the payloads into the cargo bay at the launch facilities.

The number of payload/Orbiter cable interfaces varies widely with the various payloads and in most cases is expected to be large enough to cause potential problems in cable routing, data accuracy, and electrical interference between adjoining cables. The use of an MDM(s) by the payload would permit the connection between the payloads and the Orbiter to be a data bus. This MDM/data bus method of data and command transmission would greatly reduce the interference, accuracy, and space problems. The design of a modular-type MDM (smaller, reduced capability versions of the Orbiter MDM) as MMSE would eliminate the need for the individual payload disciplines to procure their own MDM-type of equipment.

To take full advantage of the Remote Manipulator System (RMS) a variety of end effectors ("hands" which fit on the wrist of the RMS) are required as MMSE. They are to be capable of being changed, without EVA, while on orbit and will be designed to perform the various operations anticipated during routine on-orbit payload deployment, retrieval and maintenance and during emergency conditions.

The need for an EVA tool kit is obvious. The desirability and value of routine EVA as shown by the recent Space Division EVA study and the highly successful and extremely valuable emergency EVA accomplished at the beginning of the Skylab program ensure the need for a well-planned and well-designed EVA tool kit.

The MSFC- and KSC-identified MMSE requirements along with the seven new items described above were categorized in accordance with the Shuttle program Work Breakdown Structure (WBS). A new category "No WBS Category" was added as it soon became evident during the categorization process that many of the flight requirements did not logically belong in any of the present Shuttle program WBS categories.

Figure 2-2 shows a portion of the table which lists the Task 1 requirements in accordance with the Shuttle WBS. The completed table can be found as Appendix A2. The left-hand column of these tables is the SD identification number for the requirement and is made up of the abbreviation for the peculiar WBS category and a requirement number (e.g., R1) assigned sequentially within the WBS category. The second column on the left is the identification number assigned in the past MMSE study with the third and fourth columns showing the page reference to the MMSE study final documentation.

The columns to the right of "Requirement" are part of the Task 2 effort and will be discussed later.

ORIGINAL PAGE IS OF POOR QUALITY

SD ID	MMC MMSE REFERENCE: ID	MMC MMSE REFERENCE: VOL II PAGE NO.	MMC MMSE REFERENCE: VOL III PAGE NO.	REQUIREMENT	POTENTIAL STS/OTHER APPLICABILITY: AS IS	POTENTIAL STS/OTHER APPLICABILITY: MOD	POTENTIAL STS/OTHER APPLICABILITY: NO	DISPOSITION RATIONALE & COMMEN
				1.3.1 STRUCTURES (ST)				
				1.3.2 PROPULSION (PR)				
				1.3.3 POWER				
				1.3.3.1 ELECTRICAL POWER GENERATION (SP)				
EP-R1		64	9, 30	AUXILIARY POWER UNIT (TUG & IUS)	X	-	-	LOOK AT LOW COST BATTERY OR BATTERIES FROM OTH ADDITIONAL BATTERIES CAN BE ADDED TO THE IUS P
EP-R2		SD REQUIREMENT		3 AUXILIARY POWER SYSTEM	-	-	-	WILL BE DESIGNED BY SD AS A MISSION KIT
				1.3.4 AVIONICS				
				1.3.4.1 GUIDANCE, NAVIGATION & CONTROL (GN&C)				
GN-R1		36	3, 33	2 SMALL IPS (MINIATURIZED POINTING & MOUNT)	-	X	-	GODDARD SIDS MAY BE CANDIDATE (BASED ON O
GN-R2		36	4, 33	1 BOOM PLATFORM	-	-	X	NOTHING EXISTING IN STS PROGRAM - BUT SOMETH
GN-R3		36	4, 36	INERTIAL MEASUREMENT UNIT (IMU)	-	-	X	ORBITER IMU DRIFT RATE IS 35 TIMES GREATER
GN-R4		36	4, 34	CELESTIAL SENSOR-COARSE (10-30 ARC SEC)	-	MC431-0128		SHUTTLE TRACKER CAN BE MODIFIED WITH LENS S UP TO 1 ARC SEC & 0.1 ARC SEC STABILI
GN-R5		36	4, 35	CELESTIAL SENSOR-FINE (0.5-1.0 ARC SEC)	-	MC431-0128	-	
GN-R6		36	6, 36	EARTH (HORIZON) SENSOR (180-360 ARC SEC)	MC432-0214	-	-	GPS ITEM HAS SUFFICIENT ACCURACY TO SATIS
GN-R7		36	6, 37	1 SOLAR SENSOR (180 ARC SEC)	-	-	X	GPS SUN SENSORS MC432-0216 & MC476-0143 A
GN-R8		36	6, 37	SOLAR SENSOR (0.5-1.0 ARC SEC)	-	-	X	GPS SUN SENSORS MC432-0216 & MC476-0143 AR
				1.3.4.2 COMMUNICATIONS & TRACKING (CT)				
CT-R1		91	10, 42	TV CAMERA (1024 LINES)	-	-	X	NO CAMERA WITH THIS RESOLUTION IN STS PROG STS EQUIPMENT. NEW DESIGN REQUIRES MONITO
CT-R2		91	10, 42	TV CAMERA (COMMERCIAL)	ICD 3-0050-01	-	-	ORBITER TV CAMERA - 525 LINES/FRAME CAN B
				1.3.4.3 DISPLAYS & CONTROLS (D&C)				
DC-R1		82	10, 41	2 PAYLOAD SPECIALIST STATION (PSS)	X	-	-	MISSION KIT TO BE DESIGNED BY RI WILL SA
				1.3.4.4 INSTRUMENTATION (IN)				
IN-R1		91	16, 42	PROTECTIVE DEVICE - EARTH/MOON/SUN SENSOR	MC431-0128	-	-	THE PROTECTIVE DEVICE USED IN CONJUNCTION
IN-R2		91	16, 42	PROTECTIVE DEVICE - RADIATION DETECTOR	-	-	X	NO SUITABLE DEVICE ON STS PROGRAM
				1.3.4.5 DATA PROCESSING & SOFTWARE (DP)				
DP-R1		69	10, 40	MINI/MIRCO COMPUTER	X	-	-	THERE ARE NO STS SMALL (800 WORD MEMORY) COMPUTER IS UNDERWAY AT AUTONETICS (REPO
DP-R2		SD REQUIREMENT		3 PAYLOAD MULTIPLEXER/DEMULTIPLEXER	-	MC615-0004	-	MDM WILL BE MODIFIED TO PROVIDE A MODUL
				1.3.4.6 ELECT. POWER DIST & CONTROL (PD)				
PD-R1		53	6, 38	REGULATOR - 28 ±1% VDC	-	-	X	±2% IS AVAILABLE, BUT NO ±1%
PD-R2		53	6, 37	DC/DC CONVERTER - 5 VDC	-	-	X	THERE IS NO SEPARATE DC/DC CONVE

Figure 2-2. MMSE Airborne Requirements/STS Applicability (EXAMPLE)

Four additional "soft" requirements were also identified and are listed in Table 2-2.

Table 2-2. New "Soft" MMSE Requirements

1. Floating pallet
2. Variable/multi-voltage power conditioner
3. EMI detector
4. Installed payload ground cooling unit

These items are not truly requirements for MMSE at this time as further investigation into the actual payloads is necessary to verify the need, by more than one payload, for each of the items. However, there are sufficient indications as to their need at this time to justify their identification as "possibles".

TASK 2 DESIGN ANALYSIS

Each of the 204 requirements from Task 1 were individually checked against the available STS/other equipment to determine the degree of applicability. The data sources for this review were: (1) results of the IR&D in-house study "Shuttle/Other Program Equipment Characteristics" for the airborne equipment which includes lists of equipment from the Orbiter, Global Positioning Satellite (GPS), Transtage IUS, and Spacelab programs, and (2) the GSE Model Review Status--Design Development and Evaluation" 2604-01-075 for the Shuttle program ground equipment.

In the baselined KSC MMSE study the GSE requirements were split into three major categories, A, B, C, and a fourth which included many pieces of equipment which were thought to be MMSE at the beginning of the study but were, in the final report, deemed not to be MMSE and dispositioned into four sub-categories.

In this study each of 120 GSE requirements, regardless of its category in the previous study, was compared to the available STS/other program equipment and potential matches were identified. In many cases, more than one piece of equipment was found to satisfy a requirement. A piece of equipment (end item of GSE) was said to satisfy a requirement if either it was deemed to be applicable "As-Is", meaning no modifications were required, or "Mod" which meant minor modifications would be required before the end item would match the requirements. The WBS for the GSE showed only one category consistent with the same level of detail for the airborne equipment. However, a breakdown of the GSE was made by the type of equipment; namely, auxiliary, checkout, handling, packaging and transport, and finally, servicing. The outputs of this task reflect this categorization. The requirement/equipment comparison process showed 60 of the 120 requirements could potentially be satisfied by over 100 pieces of Shuttle program equipment. Two examples of the multiplicity of applicable GSE are (1) four different Orbiter air conditioners (S70-0573, S78-0108, S70-0202, and S70-0708) each being developed for a different portion of the Orbiter, will satisfy the need for an environmental conditioning unit (KSC MMSE item KMA-MH-44), although two of each will be required. The second example is the requirement for a multi-purpose sling set. There are 57

different sling sets listed in the Shuttle program inventory of GSE and these will surely satisfy this requirement. In several cases, a piece of equipment identified in the list of Shuttle GSE is obviously being designed to satisfy the requirement identified in the KSC MMSE study; for example, the A70-0806 end item and the payload container requirement (KSC MMSE item KMA-MH-10).

As with the GSE, the 84 airborne requirements were compared with equipment from on-going programs. Nineteen requirements were found to be satisfied by STS/other equipment. Eleven items were from the Orbiter program while eight items were from other programs. One item from the Apollo program (probe and drogue) was found to be applicable almost "As-Is".

The results of the requirement/equipment comparison is shown on the same six tables as the requirements categorization listing discussed in Task 1. A portion of the first airborne table is shown in Figure 2-2. The columns to the right of "Requirement" show the identification number (airborne-procurement document number; ground-end item number) of the piece(s) of equipment which will potentially satisfy the requirement. The "As-Is" column means no mods, the "Mod" column means minor mods are likely to be needed, while an "X" in the "No" column means no STS/other equipment was found to satisfy the requirement. The last column on the right provides space for general information, selection rationale and/or other pertinent comments.

The higher percentage of GSE versus ASE requirements being satisfied (ASE 23 percent, GSE 50 percent) is not unexpected. In general, airborne equipment is designed for a very special application, while GSE, with the possible exception of certain handling and transport equipment, has much greater general functional capability even though designed for a particular application.

TASK 3 MMSE CONCEPT DEFINITION

A further, more detailed, comparison of the requirements with the equipment identified as a potential solution or match was performed. The detailed characteristics of the airborne hardware were obtained from the respective procurement specifications and data concerning the functional characteristics of the equipment as part of a system was obtained from the applicable Requirements Definition Documents, which are Sections 17 through 20 of the Shuttle System Definition Manual. The GSE Abbreviated Item Description (AID) document and the Kennedy Space Center Support Equipment List, TR 1287, include the physical and functional characteristics, to the level of detail currently existing for the GSE items. AID sheets have been prepared for each of the items with Space Division design responsibility, while TR 1287 includes a description of the items under NASA KSC design responsibility. In most cases, the level of detail is rather sketchy at this time; however, as the design of the hardware progresses, the detailed information will be readily available by personal contact with the design groups.

During the detailed comparison only 8 of the original 78 items were an unacceptable match; thus 70 items remain as input to the special emphasis selection process of Task 4. The original 78 items are listed in two tables, one for airborne equipment and a second for ground equipment. One page of the ASE summary is shown in Table 2-3. The complete tables are in Appendix A3. The "SD ID" requirement number is the one assigned in Task 1. The equipment "SD ID" number

Table 2-3. MMSE Equipment Summary

REQUIREMENT			EQUIPMENT			
SD ID NO	MSFC MMSE ID	TITLE	SD ID NO	EQUIPMENT ORIGIN	APPLICABILITY	DISPOSITION RATIONALE/COMMENTS
		1.3.3 POWER				
		1.3.3.1 ELECTRICAL POWER GENERATOR (EPG)				
EP-R1		AUXILIARY POWER UNIT (TUG AND IUS)	EP-H1	IUS BATTERY	AS IS	ADDITIONAL BATTERIES CAN BE ADDED TO IUS.
		1.3.4 AVIONICS				
		1.3.4.1 GUIDANCE, NAVIGATION & CONTROL (GN&C)				
GN-R1		SMALL IPS (MINIATURIZED POINTING MOUNT)	GN-H1	SIPS	MOD.	GODDARD SPS MAY BE CANDIDATE.
GN-R4		CELESTIAL SENSOR-COARSE (10-30 ARC-SEC)	GN-H4	MC431-0128	MOD.	SHUTTLE TRACKER CAN BE MODIFIED.
GN-R5		CELESTIAL SENSOR-FINE (0.5-1.0 ARC-SEC)	GN-H5	MC431-0128	MOD.	SHUTTLE TRACKER CAN BE MODIFIED.
GN-R6		EARTH (HORIZ) SENSOR (180-360 ARC-SEC)	GN-H6	MC432-0214	AS IS	GLOBAL POSITIONING SATELLITE HARDWARE.
		1.3.4.2 COMMUNICATIONS & TRACKING (CT)				
CT-R2		TV CAMERA (COMMERCIAL)	CT-H2	ICD-3-0050-01	AS IS	
		1.3.4.3 DISPLAYS & CONTROLS (D&C)				
DC-R1		PAYLOAD SPECIALIST STATION (PSS)	DC-H1		AS IS	MISSION KIT TO BE DESIGNED BY ROCKWELL
		1.3.4.4 INSTRUMENTATION (IN)				
IN-R1		PROTECTIVE DEVICE-EARTH/MOON/SUN SENSOR	IN-H1	MC-431-0128	AS IS	THE PROTECTIVE DEVICE USED WITH THE ORBITER STAR TRACKER CAN BE USED.

corresponds to the requirement number with the the "R" (requirement) replaced with an "H" (hardware). The "Equipment Origin" column lists the procurement specification (ASE) or end item number (GSE) which describes the applicable piece of Shuttle (or other) hardware. The last column notes the reason for the non-applicability of the seven items, briefly states the mods to be made to applicable hardware if required or presents any other pertinent information. Some requirements can be satisfied by more than one piece of equipment; in these cases the additional items are also listed. Several of the requirements are similar to an earlier (within these tables) listed requirement but can be satisfied by the same equipment.

The major effort within this task was to prepare a conceptual definition of the equipment selected to satisfy each of the remaining 70 requirements. This is presented in two forms. The first is an MMSE Item Description Sheet which will include the purpose, description, physical, and functional characteristics, and other pertinent data as applicable and available. These sheets also show the requirement being satisfied, the new item number, and the item number from which this item is to be made. The sheets are, in reality, "fly sheet" specifications and could be used as the basis for a more formal procurement document or end item description document. Figure 2-3 is an example of the concept sheet. (All 70 of the sheets can be found in Appendix A3.) In addition, concept drawings, sketches, block diagrams, schematics, pictures and/or any other descriptive material were prepared and accumulated for each of the 70 items. These conceptual definition packages were used as inputs for the Tasks 4, 5, and 6 effort on the special emphasis items chosen in Task 4.

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE - GROUND
ID NO. - A-H2
REQMT. - A-R2
ORIG. ID - A34-364

NAME: PAYLOAD PURGE CART

PURPOSE: TO PROVIDE A POSITIVE INTERNAL PRESSURE TO THE PAYLOAD TO MAINTAIN INTERNAL CLEANLINESS

DESCRIPTION: THE PURGE CART WILL BE A MOBILE, SELF-CONTAINED UNIT TO SUPPLY SMALL QUANTITIES OF GASEOUS NITROGEN OR HELIUM FOR INTERNAL PURGE AND POSITIVE PRESSURE. THE UNIT WILL CONTAIN GAS SUPPLIES, GAGES, VALVES, REGULATORS HOSES AND FITTINGS TO INTERFACE WITH PAYLOADS OR THE PAYLOAD CONTAINER.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

WEIGHT - 272 kg (600 LB)
FLUID MEDIA - Nitrogen
PRESSURE - 13,789 pascals (200 psi)
FLOW RATE - 5.6 m3/min (200 ft^3/min)

SIZE - 1143 x 117.6 x 914.4 mm (45 x 44 x 36 in.)

INTERFACES: REQUIRES 115 VAC GROUND POWER AND 440 VAC IF HEATER IS DESIRED

OTHER DATA:

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT:

- ADD RACKS FOR K BOTTLES
- ADD TOW BAR

REMARKS:

Figure 2-3. Example Concept Sheet

TASK 4 CONCEPTUAL DESIGN

The first step in the conceptual design process was to determine the criteria to be used to select the six special emphasis items from the 70 which remained from the matching process of Task 3. Five major considerations and one minor (lesser significance) were listed and subsequently approved by the COR. They were:

1. Does the item require early development funding?

2. Are requirements for it well-substantiated or justified?

3. Is further conceptual definition required?

4. Does it have a high potential usage rate or multiple payload applications?

5. Does the cost-saving potential look good?

6. Does it provide commonality between ground and flight applications?

The sixth consideration, while being used in the evaluation, was given considerably lesser weight than the other five.

As noted earlier in the report, the GSE requirements as identified in the KSC MMSE study, have in most cases almost no details regarding their functional design requirements. Even though the Space Division-designed GSE has, in general, progressed further into the design stage than is indicated by the AID sheets, the lack of detailed requirements make it difficult to determine the extent of the modifications required to an item. Thus, no GSE items were considered for special emphasis effort in the balance of the study. However, the high potential for applicability and the resultant cost savings of the GSE being designed for the Orbiter to satisfy payload requirements make it highly desirable to do further effort in determining the detailed payload GSE requirements. In fact, if the payload requirements can be determined to be "hard" early enough, it could be possible to incorporate at least a portion of them into the basic design of the Orbiter GSE. It may even be feasible to conduct a requirements study to expand and refine the GSE MMSE requirements at this time.

Application of the special emphasis selection considerations to the 19 ASE requirements which survived the Task 3 screening process and the four (3 ASE and 1 GSE) special interest non-STS items resulted in the six items judged "best". (These items were necessarily chosen by the exercise of best engineering judgement since no rigorous formal rating system can be devised at this time.) During the course of the study, an agreement was reached with the NASA COR for choice of the six special emphasis items along with four additional items requiring more effort than could be applied in the remainder of the study and for which separate study plans have been provided.

The four items for which study plans (Appendix A5) have been provided are:

1. RTG cooling kit
2. Orbiter/Spacelab system simulator
3. Payload integrated pointing system
4. Payload specialist station controls and displays

The six special emphasis items are:

1. Star tracker
2. Multiplexer-Demultiplexer (MDM)
3. Spin-up mechanisms
4. Payload/Orbiter electrical cables
5. Payload/Orbiter fluid lines
6. Multi-discipline auxiliary payload power system (MAPPS)

Results of the study for each of the six special emphasis items are contained in this section. Since these items are largely independent of each other, they are discussed separately and individually. However, they are brought together under the Programmatics heading wherein the overall impact of implementing the special emphasis concepts is discussed. Following this, combined conclusions will be drawn and overall recommendations will be presented.

STAR TRACKER

Introduction

Almost all stellar payloads/experiments require pointing accuracies beyond Orbiter-provided pointing. The first part of the study identified the Orbiter Star Tracker (OST) as an STS-MMSE candidate. The previous Reference 1 MMSE study recommended the Orbital Astronomical Observatory (OAO) gimbaled star tracker and a high accuracy (1 sec) tracker to be developed. Therefore, the feasibility and effectiveness of using the OST was studied in more detail, as described below, to determine if it could be a more effective MMSE item.

Objective of Task

The objective of this special emphasis item study was to determine the initial feasibility and application of the OST to stellar pointing payloads. Application to earth and sun pointing payloads was not studied.

Approach

The study included an evaluation of requirements, application/technical considerations, OST modification approach, definitions, and selection, cost schedules, and recommendations. Comparisons are made to the OAO gimbaled telescope "coarse" tracker and the new development "fine" tracker which was recommended by the previous MSFC MMSE study.

It should be noted that other "all electronic" star tracker developments were not evaluated. In particular, the new low-cost star tracker under development by the Jet Propulsion Laboratory (JPL) using a charge-coupled device as the star sensor should be considered. If the concept proves out (it is now in early breadboard phase), its advantages may prove desirable.

Results

A summary of the evaluation details and considerations is presented below.

Payload Pointing Requirements

Payload pointing data was taken from the prior MMSE study, which states that the source was the October 1974 Space Shuttle Payload Descriptions (SSPD) compiled by MSFC. The July 1975 SSPD evaluation indicates about the same range of requirements and frequencies. However, it was noted that 1 arc-sec and 5 arc-sec trackers will now be needed in 1980 instead of 1982 and 1981 respectively.

The prior MMSE study data for stellar pointing payload equipment is regrouped in Table 2-4. Each experiment equipment item needing pointing (telescope) antenna, etc.) is listed. It is evident that some experiments on the same payload/pallet/platform may be able to share the same star tracker. However, detailed payload integration and design is needed to accurately determine where this can occur. Such uncertainties make it difficult to estimate total tracker needs. However, the tracker configurations for accuracy ranges needed are felt to be adequately established by Table 2-4.

Applications Considerations

Figure 2-4 illustrates the basic considerations for the Orbiter capabilities in payload pointing. The simplest approach is to completely rely upon Orbiter pointing capabilities. However, while the Orbiter can track stars to 60 arc-sec accuracy measured at its navigation base, the RCS system typically controls to ±1/2 arc-deg, with 0.1 arc-deg or better attainable at the expense of exponentially increasing RCS propellant consumption. When the IMU does not receive star updates, drift rates of about 0.01 arc-deg/hr may be of concern to some payloads. The largest pointing uncertainty, however, is the up to 1.5 arc-deg differences between the navigation base and the aft payload bay due to structural distortion from thermal and other effects. In addition, further uncertainties can occur through the payload/pallet structure. Therefore, greater than ±2 arc-deg total uncertainties can exist between the navigation base and an instrument orientation. At most, one stellar experiment from Table 2-4 could be satisfied with ±2 arc-deg. pointing errors.

The uncertainties caused by Orbiter/pallet structure can be eliminated by placing a payload star tracker at the payload location, causing the Orbiter to point with respect to that device rather than the navigation base. However, the RCS deadband characteristics still apply. Reference to Table 2-4 indicates that about 40 percent of the experiments could be satisfied with 0.5 arc-deg Orbiter control deadband and about 60 percent with a 0.1 arc-deg deadband.

The remainder of the experiments must be pointed with a more precise control system such as gimbaled or isolated platforms that have self-contained reference sensors and control loops.

The feasibility of using an OST or modified OST (MOST) also depends upon the control/reference loop mechanization. Figures 2-5 and 2-6 illustrate the same stabilized platform mechanized with a mechanical gimbaled tracker and with OST-type trackers. The gimbal tracker can provide 3-axis attitude reference for the controlling IMU by alternately scanning two stars. Three-axis stabilization is needed to provide accurate pointing unless the star tracker is boresighted on the target, for which a gimbaled tracker is not suited. Resultant 3-axis attitude determination accuracy depends on the separation between the

Table 2-4. Payload Pointing Requirements Evaluation

	ACCURACY (ARC-SEC)												
	1	1 - 1.1	2 - 4.9	5-9	10-19	20-29	30-59	60-170	180-350	360-1700	1800-3500	3600-7000	7200-
STELLAR TARGETS													
EXPMT FLIGHTS	0	10	2	6	6	1	2	3	1	9	16*	7	1
EXPMT TYPES	0	5	2	3	4	1	2	3	1	8	11	7	1
SOLAR TARGETS													
EXPMT FLIGHTS	0	3	0	0	0	0	0	0	4	0	1	1	0
EXPMT TYPES	0	2	0	0	0	0	0	0	2	0	1	1	0
EARTH TARGETS													
EXPMT FLIGHTS	2	2	0	0	0	1	2	0	1	20**	9	3	1
EXPMT TYPES	1	1	0	0	0	1	2	0	1	7	9	3	1

* INCLUDES 4 FLIGHTS OF 2 EACH.

** INCLUDES 3 FLIGHTS OF 900 ARC-SEC.

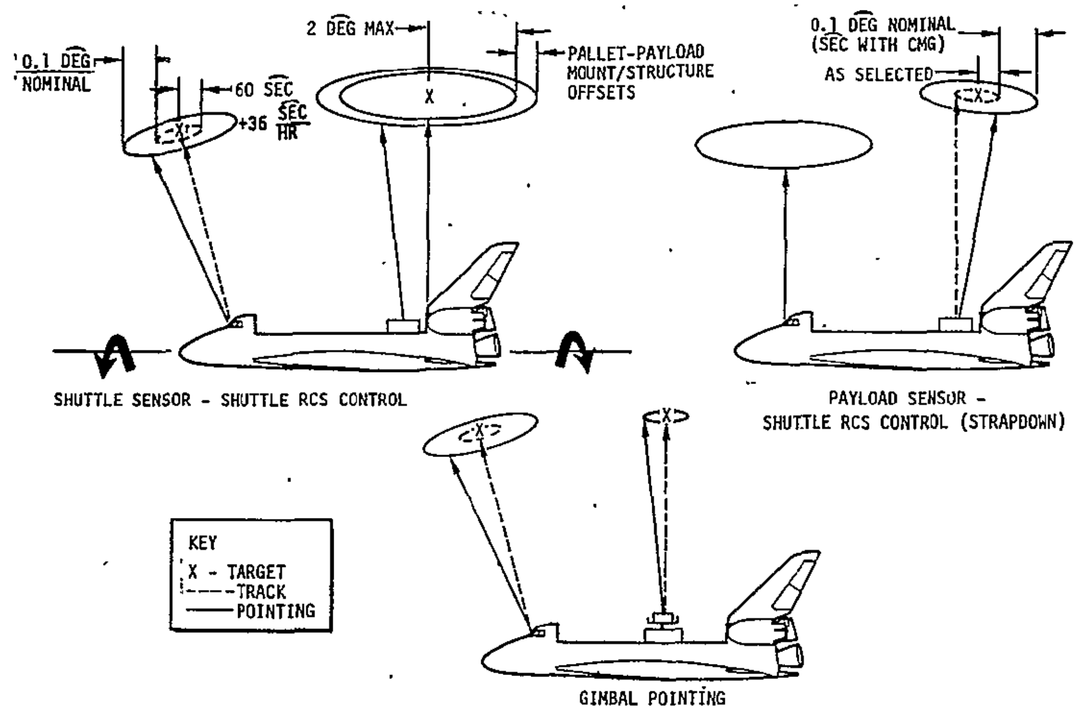


Figure 2-4. Basic Payload Pointing Concepts

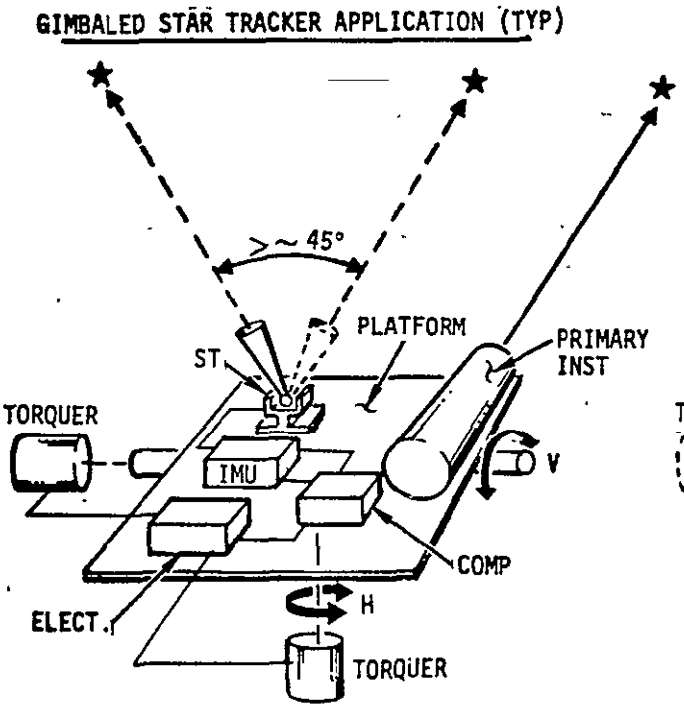


Figure 2-5. Typical Platform Mechanized with Mechanical Gimbal Tracker

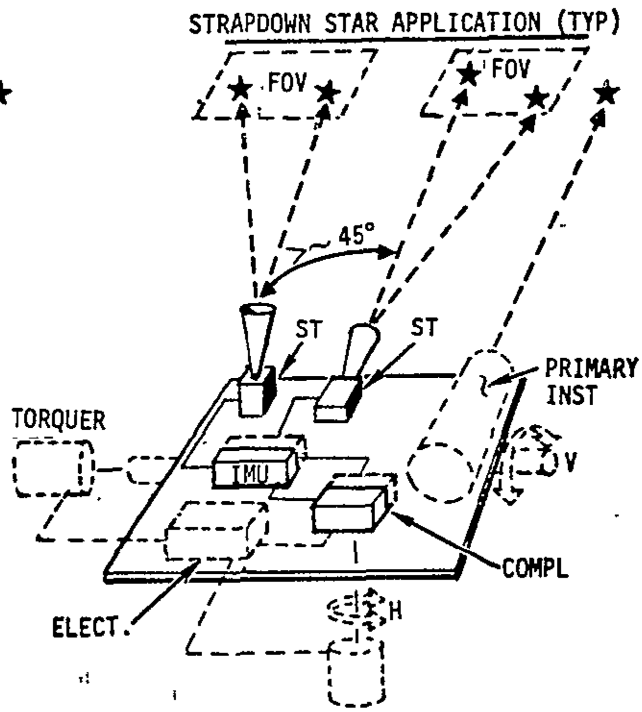


Figure 2-6. Figure 2-5 Platform Mechanized with OST's or MOST's

two stars, which should be at least 45 arc-deg with 90 arc-deg being ideal. Trackable stars near brightness +3 are adequate to assure accessibility unless combinations of earth, sun, moon, and Orbiter/payload obstructions interfere excessively.

Two MOST's are needed to achieve an equivalent mechanization to the gimbaled tracker mechanization in order to achieve the needed reference star separation, ideally oriented 90 arc-deg apart. However, unless the MOST's are mounted with respect to the primary instrument line of sight so as to assure trackable stars in their resultant FOV's, their star brightness must be increased to make use of whatever stars are available. Even so, for typical 6 to 8 arc-deg FOV's, it will be possible to find orientations where no star of necessary brightness exists. However, with a 6 x 6 arc-deg FOV and magnitude +6 star sensitivity, a probability of a star in the FOV for any 4π stellar orientation is 0.99 at average star density. Unmodified OST's could be utilized provided that mounting is such to assure +3 stars in each OST FOV for the target direction platform orientation. Or, if the IMU drift rate is low enough and initial or periodic special maneuvers are allowed, the OST's may acquire stars periodically and "reset" the IMU to the star reference.

The IMU drift rate requirements will, in general, be more stringent for the gimbaled tracker due to the cycle time to track both stars. The MOST mechanization will typically provide continuous tracking star position outputs that are updated at about 10 times per second. The IMU and associated computer capability can be considerably reduced and in cases eliminated. A system pointing study is needed to determine the precise IMU/computer needs as a function of platform characteristics for a MOST concept.

It should be noted that some pointing missions require only line of sight (LOS) (2-axis) pointing/tracking accuracy. Rotation about the LOS is not critical with respect to accuracy. In some cases, stability may be critical but not accuracy. In these cases a single MOST or OST (if the target is bright enough in the visible spectrum) may be boresighted to the primary instrument LOS. Since the OST has capability for offset tracking, taking an average of 1/4 second to acquire an offset star, a second star could be tracked to provide a relatively coarse reference for rotational control. Figure 2-7 illustrates the level of accuracy attainable as a function of star separation and LOS accuracy. An IMU to maintain orientation during the short breaks would be needed.

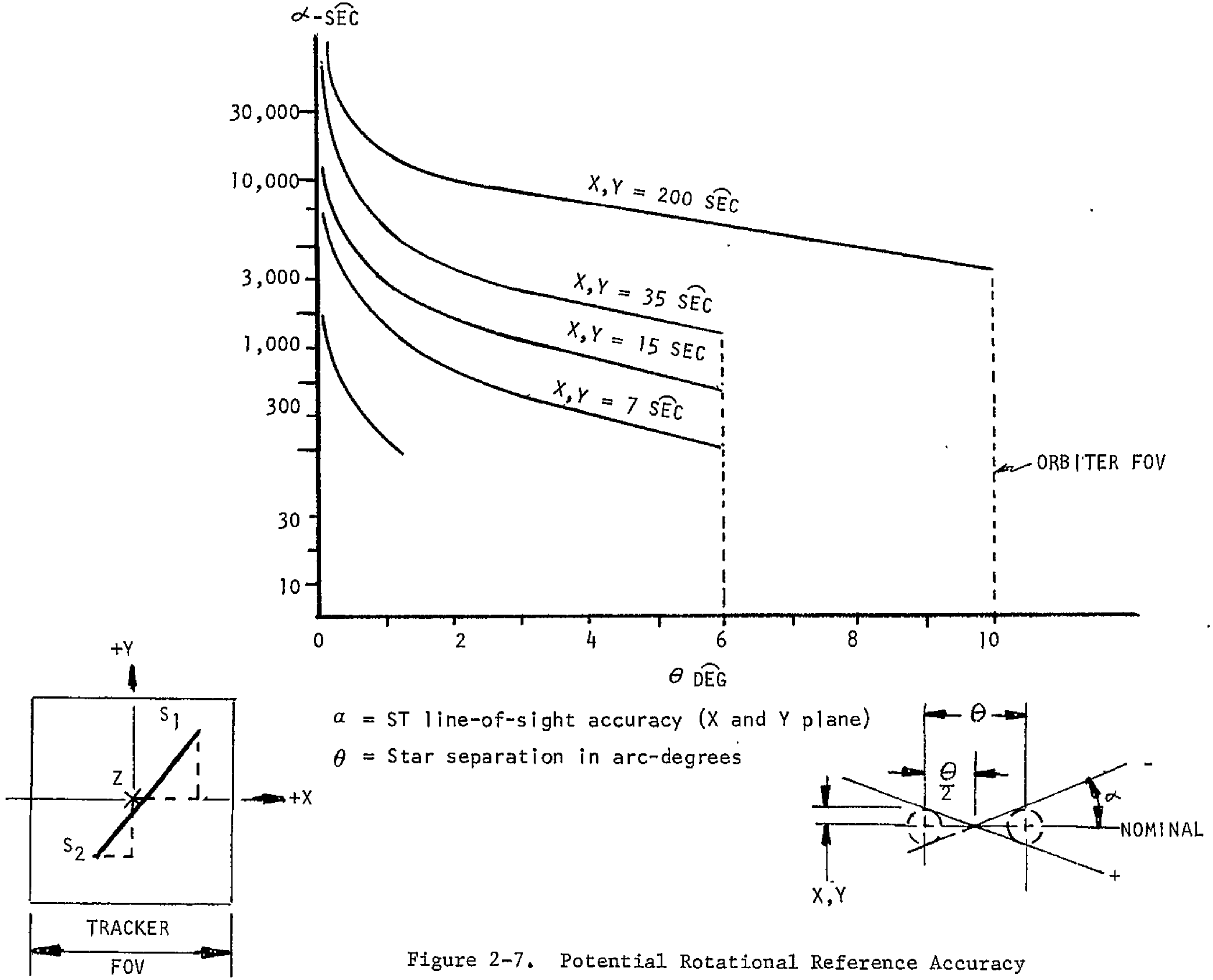


Figure 2-7. Potential Rotational Reference Accuracy

Orbiter Star Tracker Description

This section describes the OST to provide background and insight for the modifications to be described later. Figure 2-8 shows the Orbiter star tracker and its matched lens shade. Table 2-5 summarizes the tracker and shade characteristics and capabilities as pertinent to this discussion. The tracker is completely self-contained, with necessary electronics, in the one housing. The tracker and light shade mount on opposite sides of a precision planar mounting base which becomes the attitude reference for the vehicle/platform in which the tracker is installed. The light shade has an attached bright object detector (BOD), and in the lower throat of the shade, a motor-driven shutter. These protect the tracker image dissector tube (IDT) from damage due to excessive light from the sun, earth, moon, reflections, etc., that may enter the tracker field of view. The total weight of the tracker and shade is about 18 pounds. The only inputs required to obtain tracking output to better than 60 arc-sec accuracy with magnitude +3 stars are raw 28 vdc power and commands.

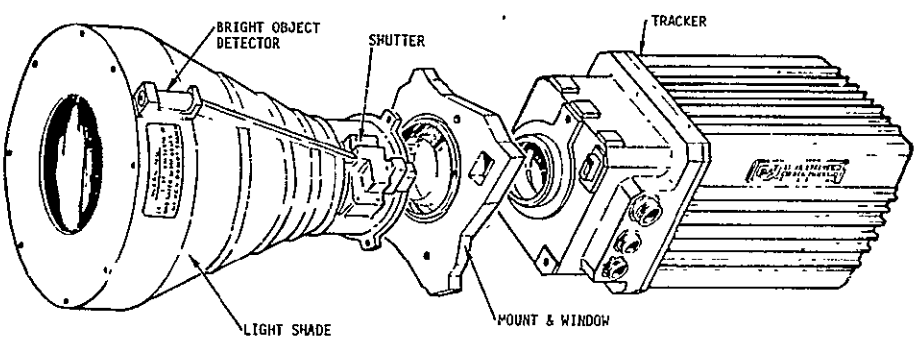


Figure 2-8. Orbiter Star Tracker and Light Shade Assembly

Figure 2-9 is the OST block diagram., Figure 2-10 shows OST internal construction. Target light enters the OST IDT via the lens assembly and the light shade, which attenuates unwanted side lighting. A photocathode in the IDT focal plane FOV emits electrons at a rate proportional to target photon rates. The electrons emitted over the FOV are accelerated to a plate containing a central pinhole, under the control of a focus coil and vertical and horizontal sweep (magnetic deflection) coils. The IDT focal plane FOV emissions are swept horizontally and vertically in increments such that the pinhole is effectively swept over the FOV in a back and forth raster pattern until a target is sensed. Then a "figure 8" local scan pattern tracks the star horizontally and vertically ten times per second. The stars centroid position is electronically determined and the sweep deflections are adjusted to keep the star "centered". A measure of the deflection coil currents at the star centroid (actually the

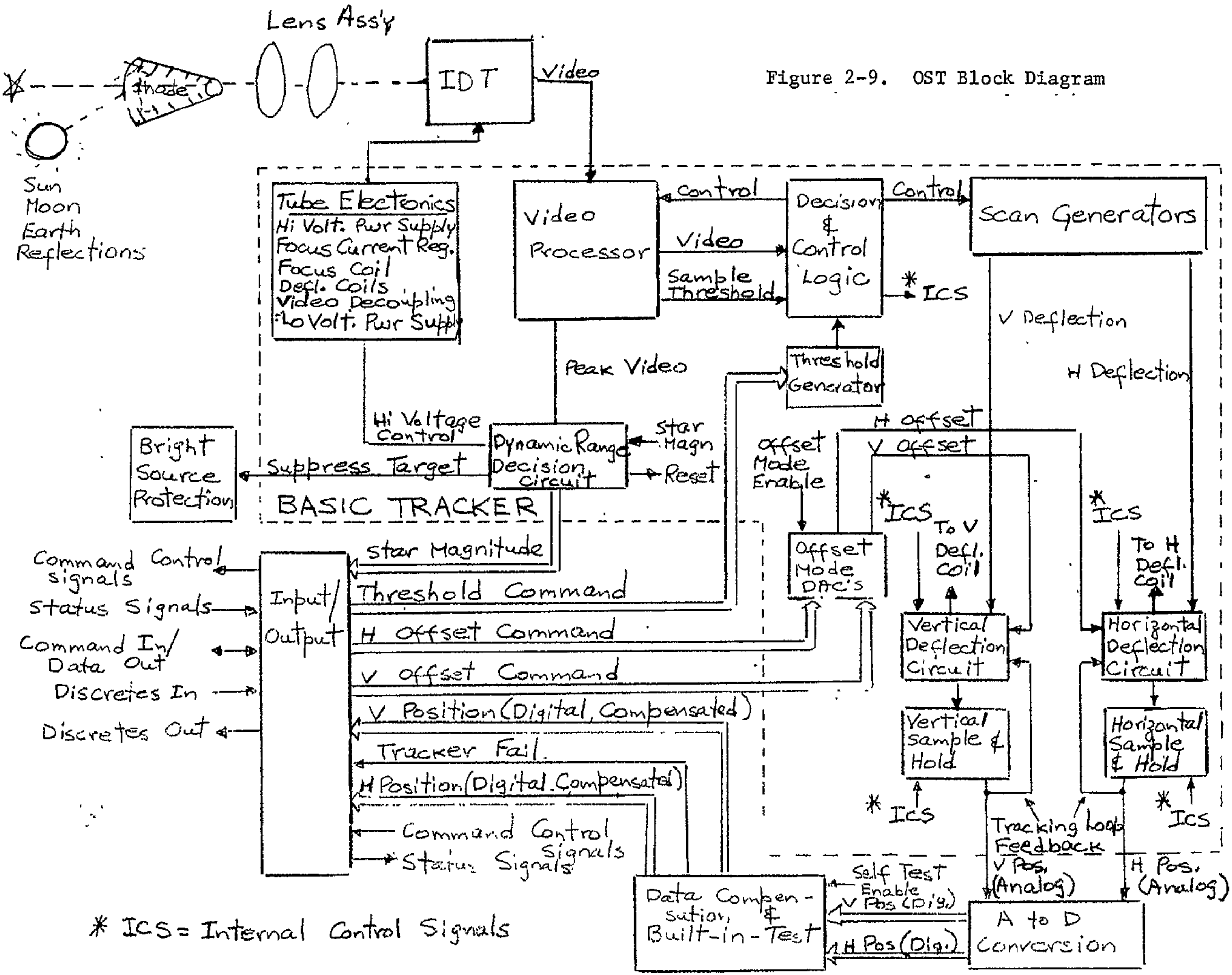
Table 2-5. Orbiter Star Tracker and Light Shade Characteristics

OVERALL SIZE (LESS CONNECTOR, VENT PROTRUSIONS) (INCHES, APPROX.)	
TRACKER	6.6 X 7.1 X 11.6
LIGHT SHADE	11 D X 11 L
WEIGHT (POUNDS, APPROX)	
TRACKER	15.2
LIGHT SHADE, WINDOW AND FRAME	2.8
GIMBALS	NONE*
LENS	56-MM FOCAL LENGTH
FIELD OF VIEW (INSTANTANEOUS)	10 X 10 ARC-DEG
STAR DETECTION SENSITIVITY	+3 MAGNITUDE (S-20)
POSITION ACCURACY (RANDOM + FIXED)	o 54 ARC-SEC
RANDOM (2-AXIS, 1σ)	31.6 ARC-SEC (RSS)
TEMPERATURE	13.0 ARC-SEC
EARTH MAGNETIC FIELD	14.4 ARC-SEC
NOISE EQUIVALENT ANGLE	13.6 ARC SEC
LAG ERROR	9.0 ARC-SEC
MECHANICAL STABILITY	3.0 ARC-SEC
LENS STABILITY	10.0 ARC-SEC
VARIATIONS IN STAR INTENSITY	8.7 ARC-SEC
VARIATIONS IN INPUT VOLTAGE	4.0 ARC-SEC
DIGITAL RESOLUTION	7.5 ARC-SEC
CALIBRATION EQUIPMENT ERROR	10.0 ARC-SEC
FIXED FIELD ERRORS - CALIBRATION ERROR	20.0 ARC-SEC
MOUNTING BASE ALIGNMENT ERRORS (TYPICAL)	o 10.0 ARC-SEC
SUN EXCLUSION ANGLE	o 30.0 ARC-DEG
EARTH EXCLUSION ANGLE	o 20.0 ARC-DEG
OFFSET SEARCH FIELD, COMMANDABLE	o 1 X 1 ARC-DEG
BUILT-IN TEST	o PRECISION ARTIFICIAL STAR IN FOV
USEFUL LIFE**	o 100 MISSIONS
STAR POSITION UPDATE RATE (2-AXIS)	o 10 PER SECOND
SEARCH ACQUISITION TIME	
FULL FIELD OF VIEW	o 5 SEC NOM; 10 SEC MAX
OFFSET SEARCH FIELD	o 0.25 SEC NOM; 1 SEC MAX

*THE ONLY MOVING PART IS THE LENS SHUTTER IN THE LIGHT SHADE.

**ONLY THE IMAGE DISSECTOR TUBE CATHODE AND SHUTTER MECHANISMS ARE POTENTIALLY LIMITED-LIFE PARTS.

Figure 2-9. OST Block Diagram



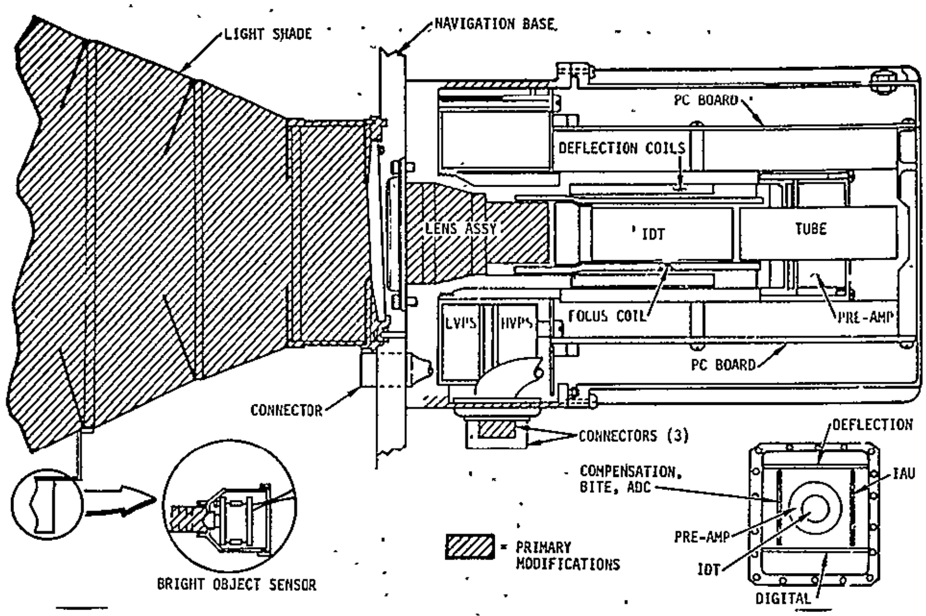


Figure 2-10. OST Internal Construction Cross Section

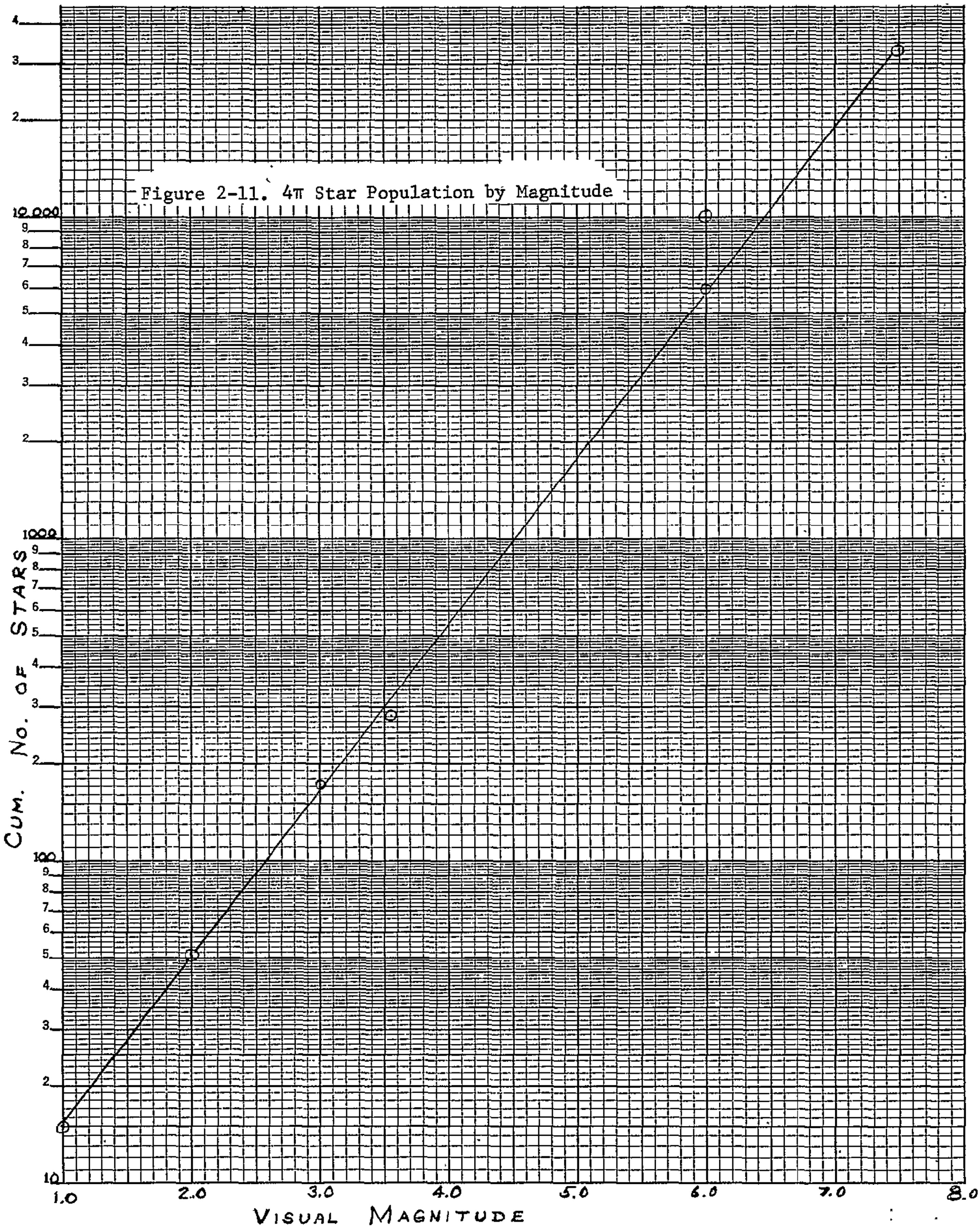
average of leading and trailing edge measurements) is a measure of horizontal and vertical position relative to a designated point in the FOV. These analog outputs are converted to parallel digital so that calibration data in a programmable read-only memory (PROM) can be applied as a function of "raw" position indication. The calibration compensates for lens and IDT distortions and mechanical/electrical fixed and temperature sensitive offsets/nonlinearities. No magnetic compensation is employed.

The IDT FOV is divided into a grid-of-squares pattern. Any analog position reading in a given square receives the same calibration incrementing so that accuracy correction near the edge of a grid square is not as precise as at the center. The finer the grid system, the more data points, the more accurate are outputs for any FOV location. The parallel outputs are not currently available at the OST external interface.

The parallel digital output is then converted to a serial digital Manchester II bi-phase level data bus code compatible with orbiter data bus. This latter step increases lag errors, which are errors due to vehicle motion during the time interval between when a control circuit receives the star position output versus when the star was actually sensed. Low vehicle rates or acceleration/velocity compensation can reduce this error. Lag and other error sources are briefly discussed below to help explain the nature of the OST modifications to be described later.

From Figure 2-11 it can be seen that there are only 157 +3 magnitude stars in 4π steradians of stellar space. This low number means that for any random strapdown tracker orientation the probability is great that no trackable star will be in the FOV. Therefore, either a pre-planned offset angular relationship between the OST and the instrument target or an IMU pointing system

Figure 2-11. 4π Star Population by Magnitude



with initial or periodic updates by OST may be required. Alternatively, the tracker can be modified for increased star sensitivity so that a trackable star is more likely to be in any FOV. Figure 2-11 indicates about 6,000 stars of magnitude +6 or brighter which, with a 6 to 8 arc-deg square FOV, provides a 0.95 probability of a trackable star in any random oriented FOV toward the galactic poles where minimum star density occurs.

Some FOV's have too many trackable stars. OST has star brightness measurement and sensitivity selectivity controls so as to ignore all but the brightness levels intended to be tracked.

OST Error Sources and Improvement Possibilities

Error sources are those due to hardware design limitations, such as mechanical-electrical offsets or instabilities, and those due to fundamental mechanization. Both are affected by the operating conditions, such as environments and vehicle/platform slew rates. Table 2-5 lists many of the major errors of concern.

Fundamental Error Sources - Lag and NEA. Fundamental errors derive from the photon statistics of stars. Compromises between star scan rates, permitted vehicle slew rate, and tracker star brightness sensitivity are needed in order to control lag error and noise equivalent angle (NEA) error with reasonable optics, and light shade parameters. This usually results in counting as few as 20 or less electrons per star detection (dwell) interval in the IDT. A variation of one or two electrons and/or their distribution during the dwell interval results in a "jitter" between successive star position readouts. The jitter can be reduced by slowing the sweep rate to allow more electrons to be generated per dwell interval. However, vehicle/platform rates must be correspondingly reduced or the lag error increases. The jitter can also be reduced by averaging several successive position readouts using a filter circuit. Here, the reduction in error depends upon the filter time constant and, again, the lag error increases for the same vehicle angular rates.

As already indicated, lag error can be reduced by reducing vehicle/platform angular motion. Application on stable platforms with small deadbands decreases motion effects. If necessary, where high accuracy is required with higher slew rates, velocity/acceleration information can be used to correct star position outputs. This can be derived from the tracker star position output change rates if accelerations or changes in acceleration are not too extreme.

Mechanical-Electrical Errors. The lens assembly, IDT, and its focus coil and deflection coils have fixed nonlinearities or distortions and variable relationships caused by thermal variations or magnetic fields. This can result in the lens assembly FOV not being linearly transformed to the IDT FOV and the electron image not being linearly transformed to the plate with the detection pinhole. The pinhole may move slightly due to thermal or mechanical effects. Fortunately, the OST design concept is inherently stable for fixed and variable errors for orbiter applications. Therefore, error sources can be corrected with calibration to a large degree. Of course, a point is reached where even small error contributions become large compared to high accuracy requirements. It appears that the present design can sustain ready compensation of errors in

order to achieve accuracies as low as 4 arc-sec, Probably to less than 1 arc-sec with special features, which need verification. Verification efforts include testing IDT internal geometry stabilities for sub-1 arc-sec performance; use of a number of built-in test stars to provide short-term calibration updates; and verification of lens/focus coil/deflection coil/IDT relationship stabilities.

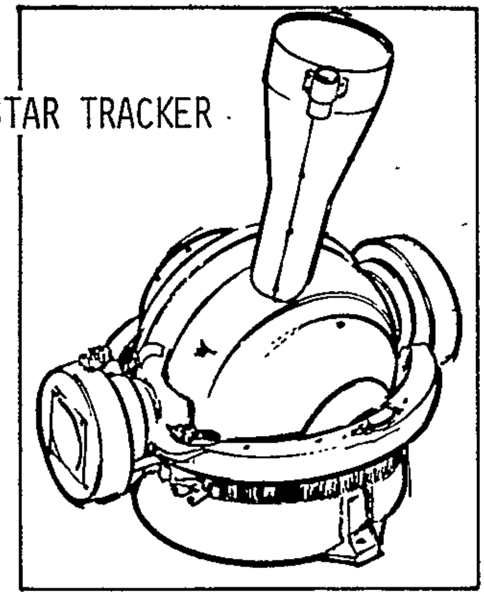
The OST outputs are compensated for thermal changes in an internal programmable read-only memory (PROM). However, the current orbiter accuracy requirement does not require magnetic compensation. Stray magnetic fields that penetrate OST shielding can exert forces on the photo cathode electrons that add or substract to those of the magnetic focus and deflection coils, causing scan position errors. This effect can be reduced to less than 0.2 arc-sec errors by incorporating a magnetic sensor to measure fields along the central FOV axis and providing a simple algorithm with calibration data storage. Magnetic effects oriented perpendicular to the electron acceleration axis (line of sight) are very small. Any added calibration (fixed, thermal, and magnetic) can be performed by external software or a microprocessor can be added internally to the tracker. Currently about 8K bits of memory are used to calibrate for thermal effects. Fixed, thermal, and magnetic calibration with double the current calibration points requires about 32K bits of memory. This can be reduced by adding a simple microprocessor algorithm to more efficiently specify calibration data storage.

Other sources of error occur in the electronics. Variations in the IDT high voltage power supply affects the velocity of photo cathode electrons; therefore, variations in time of flight. This results in different arrival points with respect to the detection pinhole. Better power supply regulation is needed at high accuracies to reduce this effect. The analog to digital converters also contribute errors. Increase in bit resolution from 8 to 10 or 12 bits is needed at high accuracy.

The calibration equipment is also a source of error. More data points require better stabilities and accuracies. To calibrate a 1 arc-sec tracker requires about 1/10 arc-sec accuracy for the calibration equipment. However, once set up, calibration is automatic with the OST concept and is of relatively small unit cost.

Mechanical Gimbal Tracker Comparison

Figure 2-12 is a sketch of the tracker used for the early Orbiting Astronomical Observatory (OAO) unmanned missions. A similar tracker was used for the Apollo telescope mount (ATM). The 2-axis gimbaled telescope uses an IDT concept similar to OST except that the target star is centered in the narrow FOV and the gimbal angles are read out with 16-bit optical encoders. Designed accuracy was 22 arc-sec. Better accuracy requires redesign and special parts selection. Limitations to encoder least significant bit limits readout accuracy. Improvements are costly and outputs are comparatively unstable due to moving mechanical relationships and gimbal pointing control dynamics. The tracker is considerably larger than the OST and requires, in addition, an electronics box larger than the OST.



WEIGHT: GIMBALS-OPTICS - 10.5 KG (23 LB)
ELECTRONICS - 10 KG (22 LB)

SIZE: GIMBALS-OPTICS - 584 X 457 X 432MM
(23 X 18 X 17 IN.)
ELECTRONICS - 406 X 305 X 127MM
(16 X 12 X 5 IN.)

ACCURACY - 22 SEC (1σ) ($\pm$15 SEC/50 MIN STABILITY)
STAR SENSITIVITY - +2.5 MAG
GIMBAL RANGE (2-AXIS) - $\pm$0.57 DEG

TELESCOPE FOV (INSTANTANEOUS & TOTAL) - 1 DEG
SUN EXCLUSION ANGLE - 30 DEG
EARTH EXCLUSION ANGLE - 15 DEG

Figure 2-12. OAO Tracker

The gimbal tracker tracks only bright stars since it has gimbal freedom to "look for" appropriate stars. Therefore, it requires less star catalog data storage support.

The OAO tracker was recommended as the coarse tracker MMSE by the prior MMSE study. Therefore, it is assumed that the OAO tracker was the most suitable of its type. Other gimbal trackers were, therefore, not examined.

MOST Options

Consideration of the payload requirement ranges, the need to achieve increased OST sensitivity, and inherent OST error sources, all discussed previously, can lead to a number of possible MOST options. However, satisfying as many applications as possible with minimum modification is a valid first step.

Minimum Modification Case. In addition to those cases where a mission may be planned such that an unmodified OST can track a +3 star, simple electronics parts interchanges can slow the star scan rate and thereby achieve increased sensitivity, as discussed before. The above cases, in conjunction with the Orbiter IMU, should be adequate for most payload sensor, Orbiter controlling, pointing missions (40 to 60 percent of stellar experiments). The precise dynamics should be studied to completely verify this likelihood.

Table 2-6, Options 1A, 1B, and 1C summarize the MOST/OST options that may be applied. Increased calibration for Option 1C would probably be required. Some stabilized platform requirements can possibly be satisfied by Option 1C with delta calibration. However, the platform control loop dynamics and associated IMU/inertial reference requirements needs analysis to determine if the increased MOST lag time due to slower scan rates (5 per second) is a factor in cost and/or performance.

Sub-1 Arc-Sec Case. This case could conceptually fulfill the highest accuracy payload applications and perhaps also perform the pointing for missions requiring up to 15 or 20 arc-sec. However, cursory analysis from Ball Brothers indicates that MOST length will increase 6 to 8 inches due to the need of approximately an 18-inch (folded) focal length lens needed to increase star sensitivity. This decreases NEA errors. The resulting design tracks +9 magnitude stars and has a 1-1/2 arc-deg square FOV for a probability of a trackable star in the FOV of about 0.99 for average density of stars. Stray light must be attenuated greatly to discriminate against trackable stars. The shade is estimated to be 4 to 5 feet long and about the same diameter. This will require special mounting considerations and use will be undesirable except where necessary. Table 2-6 summarizes the changes and characteristics for the 1 arc-sec MOST (Option 3A). Option 3B allows sub-1 arc-sec accuracy with long integration time (increased from 1 second to 5 seconds). The lag time must be considered in the pointing mechanization.

A mechanization dynamics study is recommended to ascertain permissible lag times. Detailed design analysis is also required to optimize the compromises in lens size/focal length, light shade, track scan (dwell) time, and other error source controls for the range of applications.

While the OST design concept is inherently stable (or predictable/repeatable) with respect to error sources, and the same basic design is expected to be adequate to 5 arc-sec, 1 arc-sec accuracy potential should be validated with hardware testing. IDT/focus coil, deflection coils, electronics, lens assemblies, and mounting stabilities from launch/thermal environments should be tested and necessary techniques defined. The use of built-in calibration updates with a "field" of built-in test stars should be investigated.

5-10 Arc-Sec Tracker. Since the 1 arc-sec tracker will be too unwidely for general application except as needed; since the simplest MOST cannot reach 5 to 10 arc-sec accuracies; since a substantial number of experiments need 5 to 20 arc-sec pointing; a third MOST configuration is needed. Options 2A, 2B, 2C, 2D, and 2E in Table 2-6 are possible. The basic changes evolve from increasing lens size for greater star sensitivity, increasing output filtering for NEA jitter, and increasing calibration. Calibration can be added externally or internally to the MOST.

Table 2-6. OST Modification Options

MODIFICATION DESCRIPTION*		EXTERNAL SOFTWARE SUPPORT	STAR SENS. (MAG)	FIELD OF VIEW (DEG)	ACCURACY (ARC-SEC) (1σ)
1A	EXISTING OST WITHOUT CALIBRATION	NONE	+3.0	10 X 10	<200
1B	EXISTING OAST NO CHANGES	NONE	+3.0	10 X 10	< 60
1C	INCREASE OST DWELL TIME (MINOR ELECTRONIC CHANGES)	32K BITS/SPECIAL PROCESS. TO REDUCE BITS STORED	+5.5	10 X 10	15 (BEST)
2A	CHANGE LENS TO 95-MM & MAKE CORRESPONDING CHANGES TO LS, BOD, HOUSING; EXTEND STAR BRIGHTNESS MEAS. TIME	NONE	+6.3	6 X 6	35
$2B_1$	2A, EXCEPT ADD MAGNETIC SENSOR; ADC ACCURACY IMPROVEMENT; POSSIBLY HV REGULATION IMPROVEMENT; INCREASE CALIBRATION POINTS (NO DATA INCREASE)	32K BITS + MAG. COMP. BITS AND/OR SIMPLE PROCESSOR (SAME RESOLUTION AS CT401)	+6.3	6 X 6	7
$2B_2$	$2B_1$, EXCEPT ADD DATA PROCESSOR	NONE OR LITTLE	+6.3	6 X 6	7
2C	$2B_1$ OR $2B_2$ EXCEPT INCREASE OUTPUT INTEGRATION TIME TO 1 $\widehat{SEC}$ TO REDUCE NEA	$2B_1$ OR $2B_2$	+6.3	6 X 6	4
2D	2A, EXCEPT PROVIDE NO OUTPUT INTEGRATION	$2B_1$ OR $2B_2$	+6.3	6 X 6	14
2E	2A, EXCEPT PROVIDE NO CALIBRATION	NONE	+6.3	6 X 6	< 200
3A	CHANGE LENS TO 457 MM LENS, LS, BOD, HOUSING; STABILIZE IDT-FOCUS COIL; CHANGES FROM 2A AND 2B; ADD CALIBRATION POINTS (SIGNIFICANT DATA/EQUIP. DELTA); SLOW TRACK-SCAN PERIOD TO 0.2 SECOND	$2B_1$ OR $2B_2$	+9.3	1-1/4 X 1-1/4	4 (WITHOUT OUTPUT INTEGRATION)
3B	3A, EXCEPT WITH 5 $\widehat{SEC}$ TIME INTEGRATION PERIOD (OR INCREASE LENS SIZE)	SAME AS 3A	+9.3	1-1/4 X 1-1/4	0.8

*ALL MODS INCLUDE ANALOG AND PARALLEL DIGITAL OUTPUTS AND ABILITY TO REMOVE THE SERIAL DIGITAL MODULE.

It would be possible to use the 5 to 10 arc-sec tracker for coarser accuracy applications. However, some impact in size and cost occurs. The housing length is expected to increase by about 2 inches and the light shade will increase from 11 to about 18 inches in length.

Costs

Rough order of magnitude costs were estimated with the assistance of manufacturers. Table 2-7 lists costs for each of the three selected MOST options and for the ATM gimbal tracker which is similar to the OAO tracker.

Table 2-7. Cost Data on Alternative Star Tracker Design Variations

STAR TRACKER CHARACTERISTICS	GIMBALED SKYLAB ATM	ORBITER STAR TRACKER CHANGE CATEGORY		
		I (BASIC)	II	III
FOV (DEG.)	+87°OG, ±40°IG	10 x 10	6 x 6	1.25 x 1.25
STAR SENSITIVITY	MAG 3	MAG 3	MAG 6.3	MAG 9.3
ACCURACY	10-30 SEC	60 SEC	4-35 SEC	0.8-4 SEC
COST DATA				
NON-RECURRING				
REDESIGN & TEST (DIR)	$1,000K	--	$225K	$300K
PROG. MGMT. (40%)	400		90	120
TOTAL N-R	$1,400K	--	$315K	$420K
RECURRING (1 UNIT)				
PRODUCTION	$300K	$120K	$150K	$208K
PROG. MGMT.	90		45	62
TOTAL PROD.	$390K		$195K	$270K
OPERATIONS	SIGNIFICANT	NEGLIGIBLE	NEGLIGIBLE	NEGLIGIBLE

It is clear that the gimbal concept is not competitive given that the MOST-type trackers meet requirement, which preliminary analysis indicates that they will.

Schedules

Figure 2-13 shows recommended implementation scheduling. Reference to the July 1975 payload description documents (MSFC) indicates that payloads AS-03-S and AS-01-S require 5 arc-sec and 1 arc-sec trackers, respectively, for flights in 1980. The coarse accuracy requirement is also required in 1980.

Modifications to the 60 arc-sec MOST are minor so that about a year from purchase order to delivery should be adequate. In addition, a tracker for actual integration into a typical pointing system should not be necessary (weight, balance, and function should be easily simulated) so that delivery six months before flight should be adequate.

The 1 arc-sec tracker is the most critical since elements of design should be verified before design commitment. This should occur early enough to

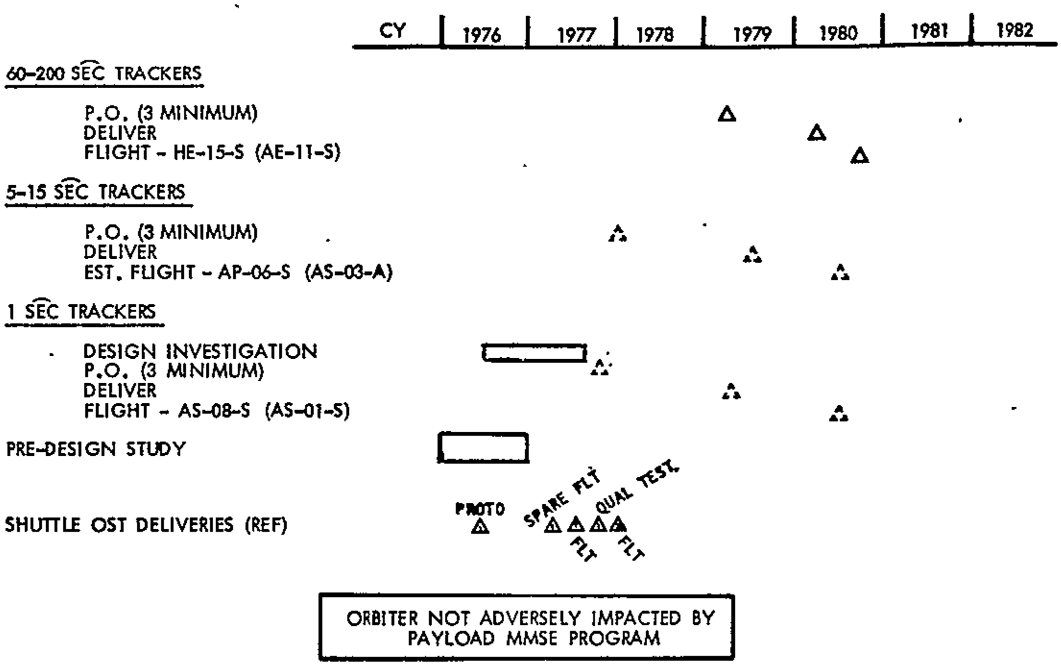


Figure 2-13. Recommended Implementation Schedules

pursue alternatives if necessary. Longer periods of time to allow design integration into the experiment and to perform integrated testing are recommended.

The 5 arc-sec tracker has less critical uncertainties in design than the 1 arc-sec MOST. Therefore, its schedule to delivery and flight can be relatively shorter.

However, as mentioned previously, analytical efforts are needed to better define and verify the exact design changes to be accomplished. Also, procurement specifications should be prepared. Therefore, an early predesign study is recommended. This study can also more specifically define the hardware design approach validation phase for the 1 arc-sec tracker.

Conclusions

A more in-depth investigation on utilizing the Orbiter star tracker (OST) as MMSE for payload applications has revealed that OST, modified, could meet the pointing requirements of virtually all sortie stellar missions. Requirements can be met by a combination of modifications and increased calibration with external support depending upon the accuracy needed. In all cases, the modifications utilize most of the existing design.

Very minor modifications are needed in the 60 arc-sec accuracy region. At the 10 arc-sec and 1 arc-sec regions, new lens assemblies, with associated changes for compatibility, and increased calibration must be added. The main modification driver is to increase OST star brightness sensitivity in order to have a high probability of having a trackable star in its field of view (FOV) for any orientation outside the sun/earth/etc. exclusion angles.

It was found that at least two, and probably three modified OST (MOST) configurations can be justified. Again, the operating principles and many subassemblies would be the same for each configuration.

It may also be feasible to utilize MMSE stellar reference for certain earth or solar pointed payloads. For earth pointing missions, either special Orbiter attitude and tracker installation must be observed (to acquire stars >20 arc-deg above the earth's limb) or an appropriately small drift-rate inertial maneuvering unit (IMU) must be used with initial (or periodic) Orbiter maneuvers to allow star tracker updates to the IMU. For solar pointing missions the Orbiter and MOST installations must be controlled to allow a tracker FOV >45 degrees from the sun. In both cases target pointing must be relatable to a stellar reference attitude. The desirability and effectiveness of the latter applications were not investigated since necessary payload information was not available.

Due to the above application uncertainties, plus the possibility for experiments/pallets to share star trackers, and uncertainty on how much time a set of MOST's are needed for a given experiment development through installation, the total procurement of each MOST configuration could not be closely estimated. Therefore, cost comparisons are not absolute regarding the effects of buy size. However, this was not a driver in determining comparative cost-effectiveness for selecting options. The selected MOST configurations were compared to the prior MSFC MMSE study tracker recommendations to verify relative cost-effectiveness.

Recommendations

A separate study is recommended to refine MOST design changes and configurations and to prepare procurement specifications. Analytical efforts to firm up precise relationships such as lens aperature/focal length/FOV/sensitivity/light attenuation relationships are needed. Also, hardware concept verification efforts are needed to evaluate the IDT and other mechanical stabilities. The use of built-in test "stars" for self-calibration in 1 arc-sec applications is another area of investigation. The study would be conducted with the participation of Ball Brothers.

In addition, the new low cost star tracker being developed by JPL should be considered before MOST procurement is initiated. The new charge coupled device sensors, replacing the image dissector tube, should simplify star "scanning" and calibration and reduce recurring costs. If development proves out the expectations, the Orbiter star tracker may not, in the final analysis, be the best MMSE star tracker approach.

MULTIPLEXER-DEMULTIPLEXER (MDM)

Introduction

Studies indicate that 500 or more signal wires may be required between the payloads and aft crew station for various kinds and mixes of payloads. Line lengths can be up to the length of the payload bay plus that at right angles as caused by routing constraints in the bay and beyond the 576 bulkhead. Long, large cables nearly always have signal losses, cross-coupling, and EMI problems that require additional signal calibration, line driver, and isolation interface hardware. Also, cable design and installation time, errors and malfunctions generally cause schedule and expenditure problems.

A means of reducing or eliminating the problems of long, large cables is to multiplex, or time share, the low information rate signals onto a digital data bus. Such data bus lines are immune to noise and loses as long as they are not so gross as to cause false state levels to occur in the digitized signal.

The Orbiter and/or Solid Rocket Booster (SRB) MDM's are logical candidates for payload use. They are reliable, flexible, and have large capabilities and capability ranges to handle Orbiter subsystem intra-communications. Development costs, although relatively large due to the capabilities, are already sunk. Production costs should be relatively low-cost due to micro-circuit technology. In addition to communication between payloads and the aft crew station, use of an MDM to interface with payload station controls and displays appears feasible.

For the above reasons, a more detailed study of the feasibility of using the Orbiter/SRB MDM's was pursued.

Objective of Task

The objective of the study effort was to determine the feasibility and cost effectiveness of applying the Orbiter and/or the SRB MDM's to payload application and to define the subsequent efforts for developing and implementing the concept.

Approach

An idea of typical payload signal characteristics and quantities that could interface with the aft crew station was required to ascertain that the MDM is suitable. The Advanced Technology Laboratory (ATL) conceptual design was the primary source of such data. The IUS and other payloads were also investigated for signal interfaces but data was less quantitative. The MDM design and capabilities were then studied and MDM capabilities were compared to the potential payload requirements. Any over or under capability was then evaluated to determine what changes might be cost effective to achieve a better matchup. Extensive coordination was conducted with Sperry-Rand (Phoenix), the MDM manufacturer, in evaluating such potential changes and estimating their costs.

Possible options were then evaluated with respect to relative/total cost effectiveness as well as technical feasibility. The impact upon the STS program due to an increased demand of common hardware was investigated. Conclusions were then drawn as to the best approach and recommendations were made to implement the concept.

Results

Requirements

The general requirement for an MDM-type unit is illustrated by Figure 2-14. Many payloads will be composed of several experiments (or spacecraft, etc.) which are controlled from the Orbiter PSS. Cables with up to 500 wires are anticipated unless multiplexing reduces the number. Long, large cables always have deleterious effects upon signals. Steps to counteract these effects require such as power/driver/isolation/buffer amplifiers and calibration hardware. End-to-end calibration cannot generally be finalized until the final integration takes place. Large numbers of wires can lead to interface matching errors. Cable installation and design can also be difficult. This affects the time for final integration and the certainty that payload elements will perform as they did at the development site without last minute fixes.

The uncertainties associated with long, large analog cables can be resolved by digitizing the information and reducing wiring through multiplexing. This advantage was recognized in designating the MDM for use in the Orbiter.

A specific requirement for an MDM-type device derives from the requirement to provide large numbers of signal paths and interfaces between the PSS and the payload bay and recognition of the problems with such requirements. The best solution to the basic requirement is an MDM-type approach for signals with which it is compatible, which are all but those that must be hardwired or those with high information rate characteristics.

MDM Description

The purpose of the Orbiter and SRB MDM's is to multiplex a large number of relatively low information rate signals from subsystems and place the time-shared result on a Manchester II bi-phase level data bus. The result is communicated to other MDM's, computers, or subsystems connected to the data bus. This reduces/eliminates long large cable problems (installation, noise, integration problems, isolation/calibration hardware). Long large cable problems will occur between the aft crew station and the upper stage, unmanned, special, and other payloads. The MDM can also accept data bus information and provide outputs to subsystems, control panel displays, etc., as readily as it sends out information. Figures 2-15 and 2-16 summarize the basic characteristics of the MDM.

A key feature of the MDM for general application potential is its input-output flexibility. Eight different input-output modules (IOM's) for discrete, analog, and serial digital inputs/outputs can be arranged in any combination to partially or fully fill the 16 IOM spaces available in the MDM. This permits any signal requirements, not exceeding the MDM total capacity, to be met

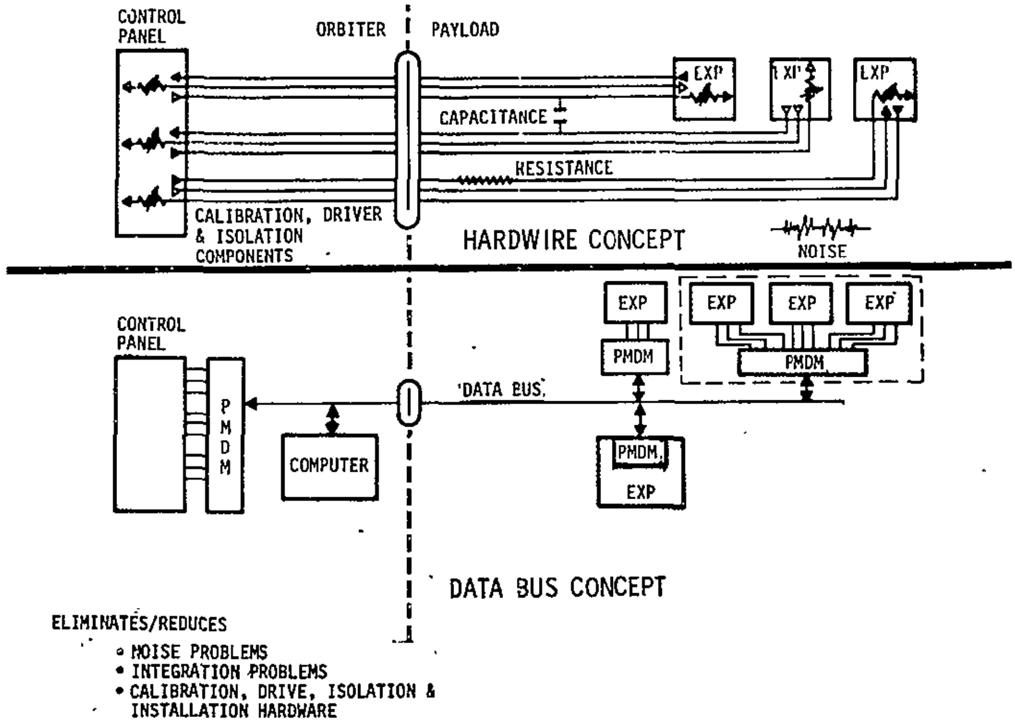


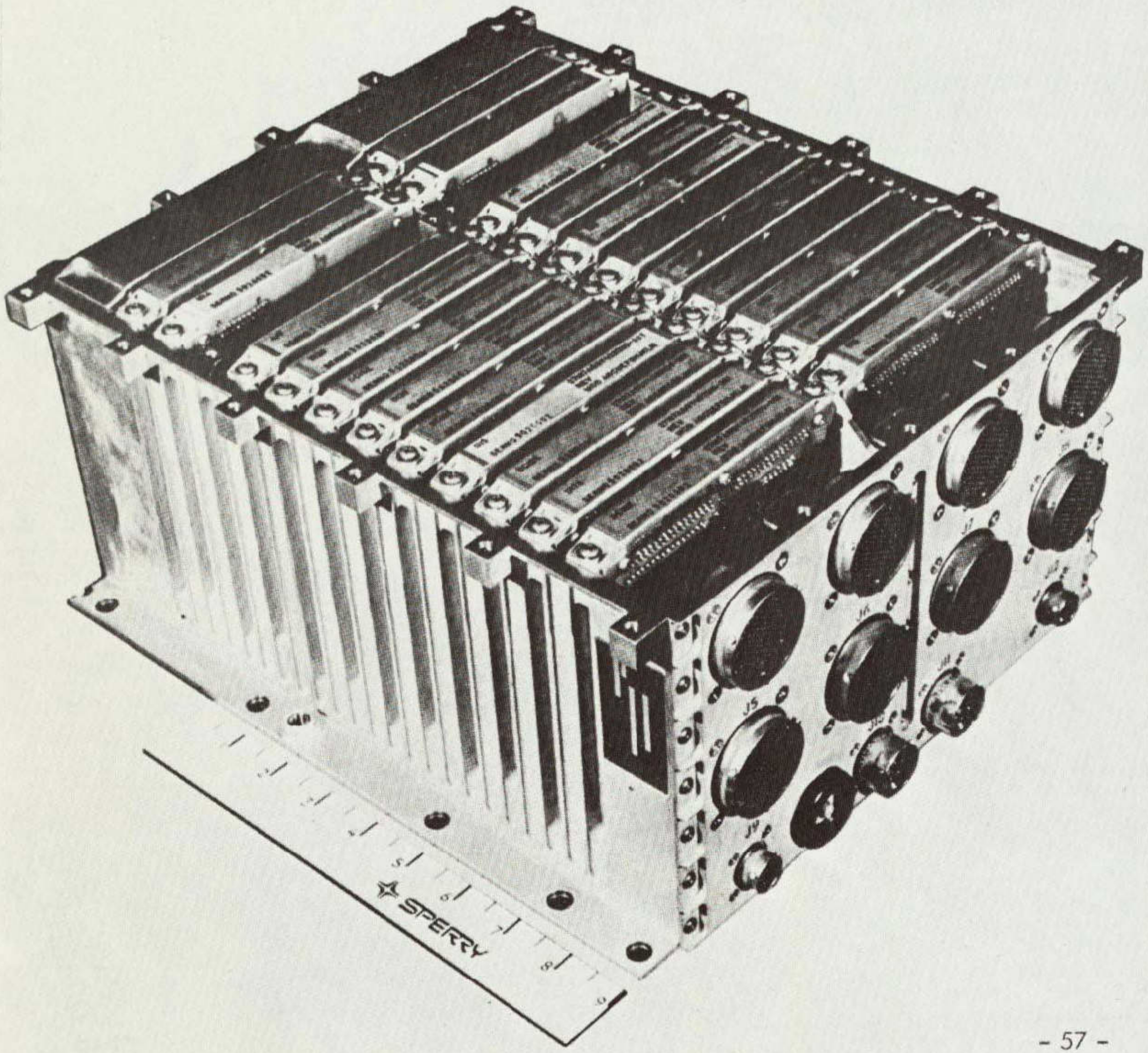
Figure 2-14. MDM Potential for Payloads

with a minimum of excess signal capacity. The fact that it communicates equally well in both directions is another key ingredient. In addition, it has fault tolerance features (signal isolation, short circuit protection, etc.) that are highly desirable; although initially relatively costly. However, development is a sunk cost for payload application.

A feature probably not necessary in many payload applications is the redundancy. Approximately 50 percent weight, size, and cost savings are possible by splitting the MDM down its length (see Figure 2-15,-16). Redesign of the motherboard, case, and connector wiring are all the significant changes required to obtain simplex (no redundancy) payload MDM's (PMDM's) with an 8-IOM capacity for input/output signals and with the flexibility, 2-way communication, signal isolation, etc. of the original MDM.

Capability Evaluation

A representative payload was mechanized to evaluate PMDM sizing and ability to meet requirements of a typical range of signal types and numbers. Inasmuch as detailed signal wires have not been generally identified by payloads, a payload being defined in-house, an ATL pallet, was used as the example. Figure 2-17 schematically shows the experiments on each pallet. The resulting signals (types and numbers) at each pallet and at the PSS are matrixed below the applicable station. Other payloads, such as the low cost modular spacecraft (LCMS) and Interim Upper Stage (IUS) with spacecraft are expected to have similar types and numbers of signals for pre-separation checkout and controls but details are not yet available.



ORBITER MDM

CHARACTERISTICS

- 1 MBPS DATA BUS INTERFACE
- 30K WORDS PER SEC (16 BITS DATA PER WORD)
- BUILT-IN TEST
- FLEXIBLE INPUT-OUTPUT MAKEUP
- UNIVERSAL MODULE LOCATION SYSTEM

MAXIMUM DATA CAPACITY (PER SIGNAL TYPE)

TYPE	INPUT	OUTPUT
DC ANALOG (DIFF	256	256
DC ANALOG (SE)	512	N/A
DISCRETE (28V/5V)	768	768
SERIAL I/O	64	64

SIZE 330 X 254 X 178MM (13 X 10 X 7 IN.)

WEIGHT 16.6 KG (36.7 LB)

POWER 34 - 82 WATTS

- 57 -

Figure 2-15. Orbiter MDM

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OF POOR QUALITY

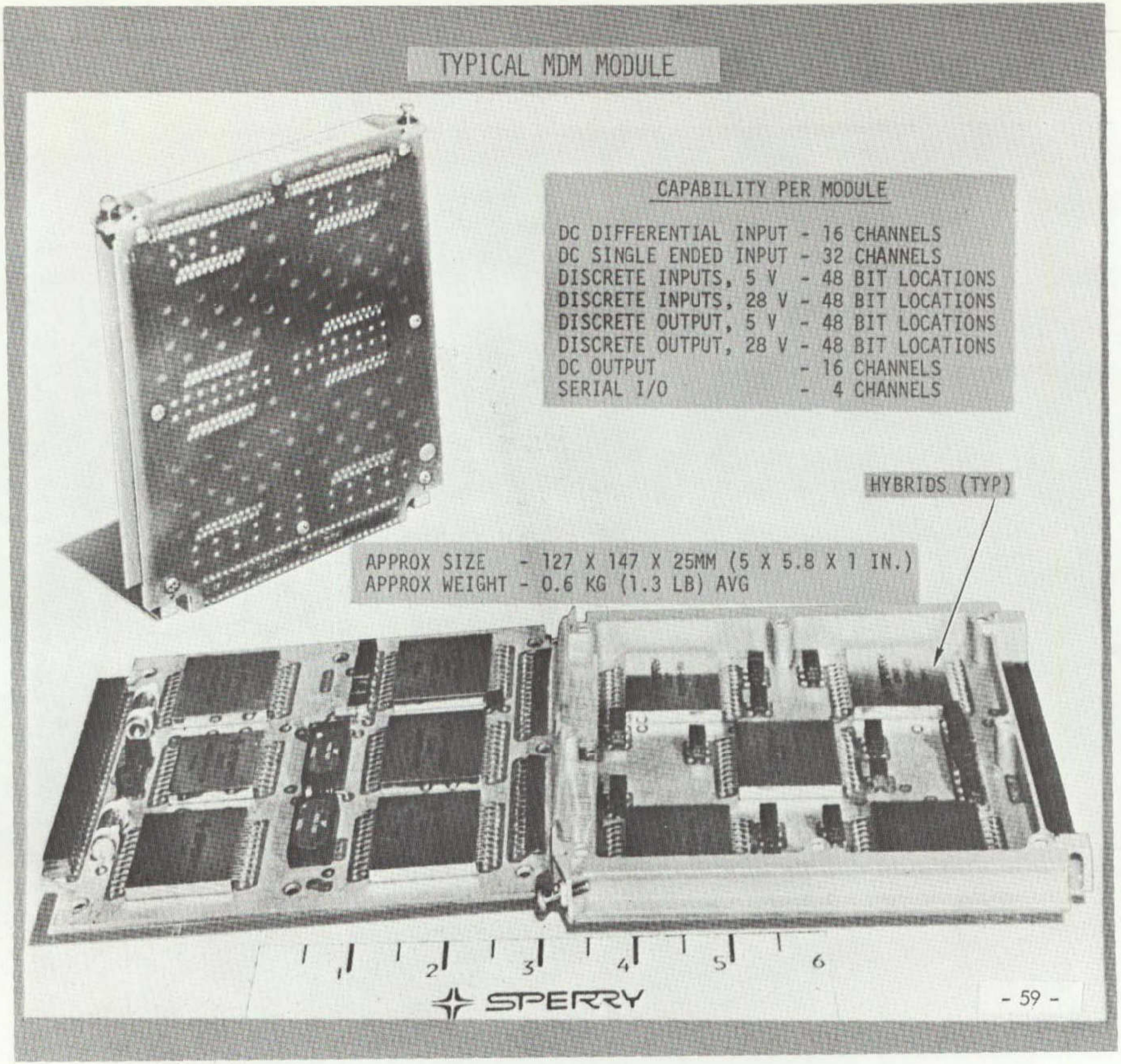
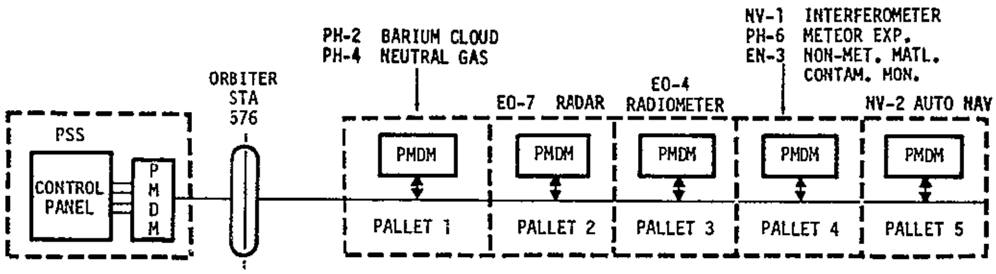


Figure 2-16. Typical MDM Module

Space Division
Rockwell International



DISCRETE INPUTS	158		26	1	1	21	21
DISCRETE OUTPUTS	37		39	20	22	44	48
ANALOG INPUTS			12	3	12	37	6
ANALOG OUTPUTS	* 25						
DIGITAL INPUTS			3	-	-	1	5
DIGITAL OUTPUTS	* 25						

* DRIVE SIGNALS FOR PANEL INDICATORS COULD BE DIGITAL OR ANALOG

Figure 2-17. Example Requirements (ATL Pallet)

Examination of Figure 2-17 shows that typical experiment groups require considerably fewer signals than a full-MDM capability. The PSS may require either a full-MDM or two "split" MDM's. A "split" MDM is one half of the full MDM to obtain single-rows of modules (see Figure 2-15). This results in a payload MDM with a simplex (no redundancy) core function (power supply, sequence control unit (SCU), analog to digital and digital to analog converter, and module interface adapter (MIA)) and space for 8 IOM's.

The IOM module count is derived from the signal capacity of each applicable signal-type IOM (see Figure 2-16). The total module count is summarized below.

PMDM IOM REQUIREMENTS FOR EXAMPLE ATL PALLET PAYLOAD

PSS	P-1	P-2	P-3	P-4	P-5
8 plus 1 to 6	4	4	3	4	5

It is seen that only 4 IOM's could handle all pallets except pallet 5. Pallet 5 needs an extra serial digital IOM to handle one excess signal since each serial digital IOM handles only four channels.

At the PSS, the number of IOM's needed depends upon the type of display indicators to be driven. If only analog displays are utilized, only one IOM over the 8 spaces available in a 8-IOM PMDM would be required (four 16-channel analog ouput modules, total, to handle the 50 analog/digital requirement plus 5 input and output discretes IOM's). If digital driven displays are assumed for the 25 channel digital requirement, then seven serial digital IOM's plus two analog output IOM's plus the five input/output discretes IOM's, or a total of 14 IOM's are required. Therefore, either one full MDM or two 8-IOM PMDM's are indicated for the PSS. However, some IOM's will have the majority of available channels unused (i.e., 158 input discretes requires four IOM's with a capacity of 192, or a surplus capability of 34 discretes, which is two-thirds of the IOM's capability). If repackaged IOM's were available that mixed types of signals, an 8-IOM PMDM could be adequate. The Solid Rocket Booster (SRB) program is developing such IOM's which are compatible with Orbiter IOM's.

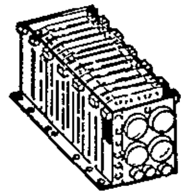
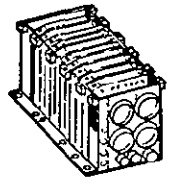
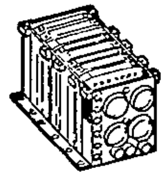
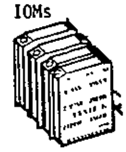
The conclusion is that an 8-IOM PMDM would handle most typical equipment groups; in fact, a 4-IOM PMDM may handle most needs. At the PSS an 8-IOM PMDM may be adequate fut a full MDM may also be needed. This is acceptable since only recurring costs apply to full MDM procurements.

To be determined is the procedure/redesign (minor) required to reprogram the Sequence Control Module (SCU) in the PMDM's. A fixed, replaceable read only memory addresses the PMDM modules based on location; therefore, needs changing when different IOM's and locations are used. The redesign must also assure adequate power supplies for worst case mixes of IOM's and provide for adequate thermal radiators/conduction away from the PMDM. These problems appear easily solvable.

PMDM Options

The PMDM options center around IOM capacity. No other feasible redesign appears appropriate except the possibility of developing a different data bus code interface module. The existing Manchester II code used by the Orbiter is advantageous unless higher information rate transfers are desired at the expense of noise/error rejection and self-test capabilities. This does not appear warranted at this time. However, such a module could be developed and merely plugged into the PMDM in place of the current MIA core module. The SCU, which handles the internal addressing of IOM's and directs analog to digital conversions, etc., could be simplified since it is designed to handle 16 IOM's and redundancy switching of core modules. However, the recurring-only cost of the delta capability is relatively small compared to a new development cost. This question warrants further study.

Therefore, the PMDM options considered differ only in package sizes (or IOM capacities). Figure 2-18 summarizes the results. It turns out that case redesign is essentially independent of size. Since the 8-IOM case can be used with only four, or even fewer, IOM's, it is the logical choice for the basic PMDM. It achieves a 50 percent size, weight, power savings over the full MDM and is only 5 in. longer and about a pound heavier than the 4-IOM PMDM, which may not handle the more complex experiment groups. Of course, a 4-IOM (or smaller) PMDM could also be developed if detailed study justifies its capacity range.

	SIMPLEX MDM A	SIMPLEX MDM B	SIMPLEX MDM C	MODULAR
				IOMs 
CORE (4 MODULES)	SIMPLEX	SIMPLEX	SIMPLEX	SIMPLEX
IOMs	8	6	4	AS REQUIRED
UNIT COST NR	280K	SAME	SAME	NONE *
UNIT COST R	60K + PROG MGMT	51K + PROG MGMT	42K + PROG MGMT	4.5K EA AVG
MAX CAPACITY DISCRETE I/O	384	288	192	48 PER IOM
ANALOG SINGLE ENDED OR DIFFERENTIAL INPUT	256	192	128	64 PER IOM
ANALOG DIFFERENTIAL OUTPUT	128	96	64	32 PER IOM
SERIAL I/O	32	24	16	4 PER IOM
SIZE (APPROX)	330 x 127 x 178MM (13 x 5 x 7 IN.)	279 x 127 x 178MM (11 x 5 x 7 IN.)	229 x 127 x 178MM (9 x 5 x 7 IN.)	127 x 147 x 25MM (5 x 5.8 x 1 IN.)
WEIGHT (APPROX) (ALL IOM'S INSTALLED)	8.3 KG (18.4 LB)	7.1 KG (15.7 LB)	5.9KG (13 LB)	0.6 KG (1.3 LB)

* INCREASES EXPMT INTEGRATION COST.
SPECIAL MODULES MAY BE DEVELOPED
AT ≈ 25K EACH

Figure 2-18. Payload MDM Options

It is also possible to use the IOM's internal to payload equipments. The serial digital IOM can digitize analog inputs and reduce wiring without the MDM core modules with proper integration. This can be useful, for instance, in reducing wiring across precision platform gimbals. The potential of this type application needs further study outside this contract.

Costs

The desirability of PMDM MMSE is shown by Figure 2-19. The cost of developing and procuring a new device with capability similar to the PMDM is shown as the upper curve. The 4-IOM PMDM was used as reference. Its cost of development and procurement is shown as the lower line. The recurring costs of both units are similar since they are at the same capability level (although not necessarily interchangeable). The development costs cannot be recovered, which are about 75 percent of the original MDM development cost since about 2-1/2 years of technology/experience accrues to lowering this cost (10 percent per year).

It is more likely, however, that the reason for a new development is for less capability/complexity than the MDM/PMDM. Still, even with a 50 percent reduction in development cost (which is a 50 percent or more reduction in complexity), well over 50 units (one-for-one basis) would be needed before the new development would become cost effective. Meanwhile, the 50 percent complexity unit does not have the flexibility and "fool-proof" features, likely resulting in more units being needed for the same job and more interface supporting hardware (scalers, converters, buffers, etc.). These extra costs would extend the cost crossover, with respect to PMDM units required, still farther out.

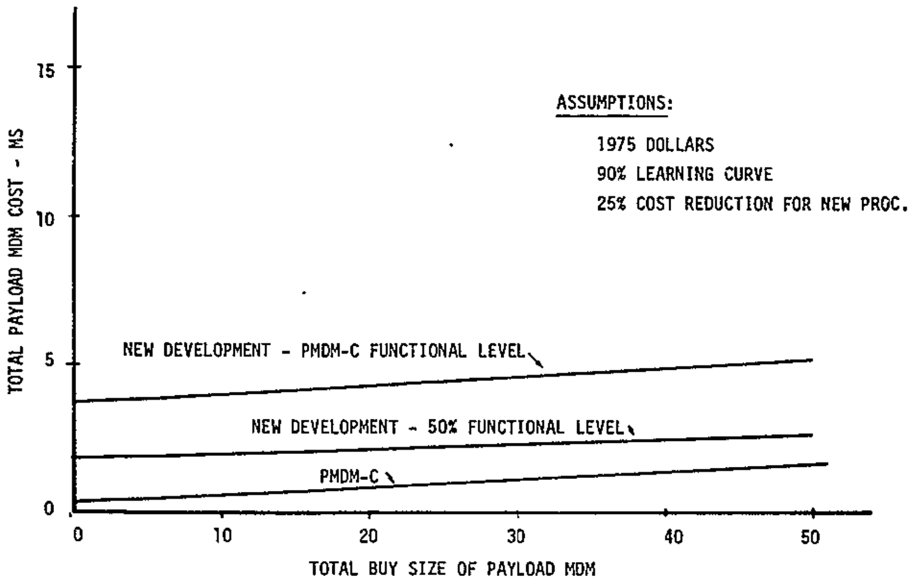


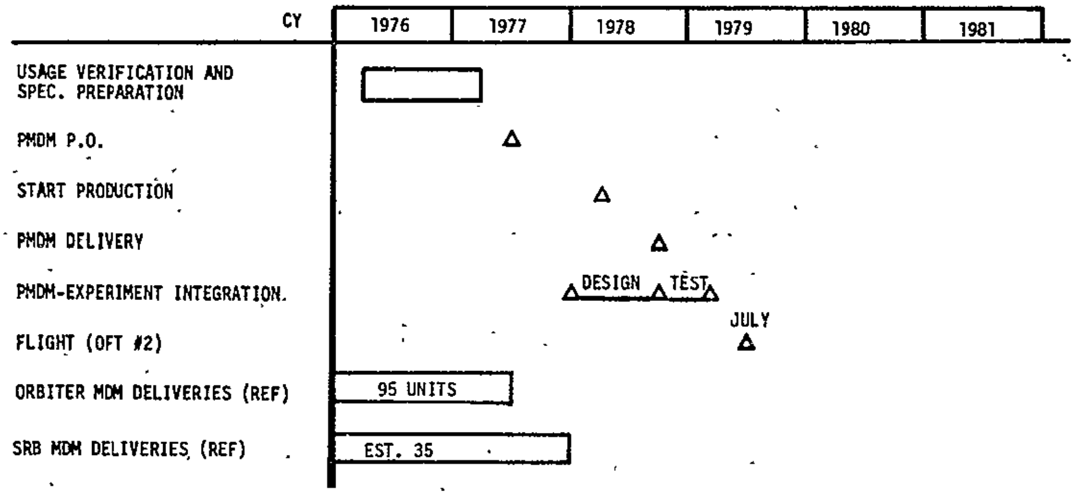
Figure 2-19. Comparative Evaluation of Payload Multiplexer Acquisition Options

While the qualitative analysis is adequate to verify the feasibility of the PMDM concept, a more rigorous scenario-type analysis might be desired. Comparisons of existing multiplexing systems, whose development costs are already sunk, with all needed ancilliary hardware/operations support would be a more rigorous approach. Such depth is out of scope of this study.

Implementation Schedules

Assuming the PMDM is to be flown early to demonstrate its usefulness to potential users, the schedule of Figure 2-20 was developed. The second Orbiter Flight Test (OFT #2) currently indicates a payload consisting of development flight instrumentation (DFI) and a group of two stellar-type experiments. The third OFT about a month later is more complex. Both appear suitable to demonstrate application of PMDM's to reduce long line, multiple signal problems with digital multiplexing. Of course, detailed evaluation of the payloads and coordination/acceptance by the responsible agencies must be obtained.

Assuming flight in July 1979, and three-fourth of a year for payload integration and experiment element developmental testing (with PMDM's incorporated to provide realistic flight data bus effects between the items and the PSS controls, etc.), first delivery is required around October 1978. Hard design information for payload designers is needed earlier. Sperry (Phoenix) estimates 15 months from purchase order to delivery, with production starting on the modified PMDM elements nine months later. Therefore, the purchase order should be let by mid-1977. Before the purchase order is let, the specific PMDM requirements, in conjunction with a representative range of users, must be determined and the procurement specification prepared. Estimating a year for this predesign activity indicates that the study should start in the first half of 1976.



CURRENT SCHEDULE INDICATES NO ADVERSE IMPACT ON STS

Figure 2-20. Preliminary Implementation Schedule - PMDM

PMDM Versus Hardwire Costs

The costs to hardwire payloads can only be roughly estimated at this time. In addition, hardwire costs compared to PMDM costs depends upon whether MMSE cables are assumed or not. Also, some hardwire cables will still be needed for high information rate signals (e.g., TV) and critical caution and warning signals. However, some insight is gained from the following. Details are covered in the Programmatics section later in this report.

The primary cost deltas between data bus and long, large cable hardwire is in the repetitive design, fab, and test of the cables, in payload bay installation and checkout/fixes, and in the interface driver/buffer/calibration circuitry in the payloads and the aft crew station. Almost all flights will require custom cable designs due to variations in payload mixes and locations in the bay. The PMDM has initial design modification costs, inventory acquisition costs, costs for a microprocessor or computer to oversee and coordinate PMDM operations, and recurring costs for computer programming and installation. Installation costs for the PMDM system and for associated buffer/driver/calibration circuitry is much less than for hardwire, however.

The result is an estimated $50M to $75M overall savings over 500 flights by using PMDM's depending on whether the comparison is against MMSE or custom hardwire. See the Programmatics section for details.

Conclusions

The Orbiter/SRB MDM's are suitable to provide 2-way data bus communications between the payloads and aft crew station (or between payloads if needed). This could alleviate signal transmission problems inherent in long, large hard-wired cables. However, less than half of the MDM capability will typically be needed at the payload interfaces. It is feasible, however, to "split" an MDM in half at the cost of a new case and motherboard. This would better match capabilities to requirements and save 50 percent of the size, weight, and costs of recurring units. The modified payload MDM would be structured with a simplex core (instead of the Orbiter/SRB redundant core) and provide 50 percent of the Orbiter or SRB IOM capability. Redundancies could still be added with reduced IOM capabilities. This approach is less costly and risky than developing a special payload multiplexer system of comparable capabilities (i.e., high error/noise immunity, 2-way single line data bus, flexible I/O, small size and weight for capability). Large overall savings are likely to occur if a PMDM concept is adopted in order to minimize hardwired long, large cables.

The unmodified Orbiter/SRB MDM's may also be usable with payload functions. Decoding and driving payload station controls and displays is a potential application. Specific payloads may find the original MDM redundancy features desirable. NASA technical personnel have proposed changing the MDM core modules to include features which would facilitate future field changes in the PMDM applicatic

Recommendations

It is recommended that plans be implemented to make payload MDM's available. However, a more in-depth study to verify that payload agencies will use a PMDM concept is recommended. More precise capability requirements can thus be developed and documented in a procurement specification. In addition, the effect on the PMDM concept due to the recent decisions to provide support provisions for a third Orbiter MDM located in the bay and dedicated to payloads needs evaluation. The need for the two payload-dedicated Orbiter MDM's at the forward bulkhead where PMDM's are located at the payloads should also be reevaluated. In order to assure PMDM support of OFT payloads, the recommended study needs to be started as soon as possible.

SPIN-UP MECHANISMS

Introduction

A type of Orbiter spin capability may be desired to deploy low altitude, spin-stabilized spacecraft. Several throw-away detectors (TAD's), AMPS subsatellites and scientific spacecraft fit this category. It was thought that a spin-up mechanism on the RMS (a type of end-effector) could provide this service. Use of the RMS for spin-up and deploy could allow more versatile and efficient stacking in the bay and minimize special deployment hardware to be developed by users. However, the RMS dynamic characteristics would be expected to limit the size and/or spin speed of spacecraft deployed by this method. In addition, RMS use would be limited to spacecraft which can orient themselves and/or are insensitive to $\sim$ 15 arc-deg deployment attitude tolerances due to the specified RMS absolute accuracy range. However, the RMS spin-up mechanism would be much simpler, smaller and lighter than the bay mounted spin-up mechanism. Determining the feasibility of an RMS end-effector type of spin-up mechanism was, therefore, one purpose of the spin mechanism study.

Another type of spin-up mechanism need was described by NASA personnel at the mid-term presentation. A substantial number of automated payloads require spin stabilization either at final orbit or during orbit transition with a kick stage. Commercial communications and other reimbursable-launch-cost payloads (44 or more from 1979-1991) may contine to use relatively small kick stages to boost spacecraft of less than 909 kg (2000 lb) to synchronous orbits (Reference 5). These use spin stabilization for kick stage orbit transfer (even though the spacecraft may be despun and 3-axis stabilized at final orbit) because this method has been the lowest cost approach to achieve synchronous orbits for such spacecraft.

The STS must provide a competitive launch capability if it is to capture the small payload high-altitude launches currently provided by Thor-Delta. One way is to assume that Orbiter will spin up and launch the same 3rd stages (spacecraft + kick stage) that Thor-Delta accommodates. This provides potential users with transition-phase benefits and simplifies cost-benefit comparisons. Recent studies confirmed the feasibility of this approach. It was, therefore, desired to evaluate previous related studies and make more specific cost effectiveness comparisons between STS versus Thor-Delta launches. This conceptual work was accomplished as a company-sponsored effort since this was beyond scope of effort in the contract study.

Objective of Task

The specific objectives of this task were to study (1) the feasibility and cost effectiveness of launching typical Thor-Delta 3rd stage payloads from Orbiter, and (2) the feasibility and cost of an RMS end-effector spin-up mechanism. If found cost effective, recommendations were to be made for program implementation steps and issues to be studied further.

Approach

Figure 2-21 illustrates the subtask logic flow. The two spin-up concepts were pursued independently and defined and evaluated to a level of detail that permitted estimates of technical feasibility, ROM costs, and implementation efforts and schedules.

The scope of the study did not permit in-depth evaluations. However, the basic conclusions are felt to be valid. Subsequent early study efforts could provide refinements and confirmation of conclusions with higher confidence.

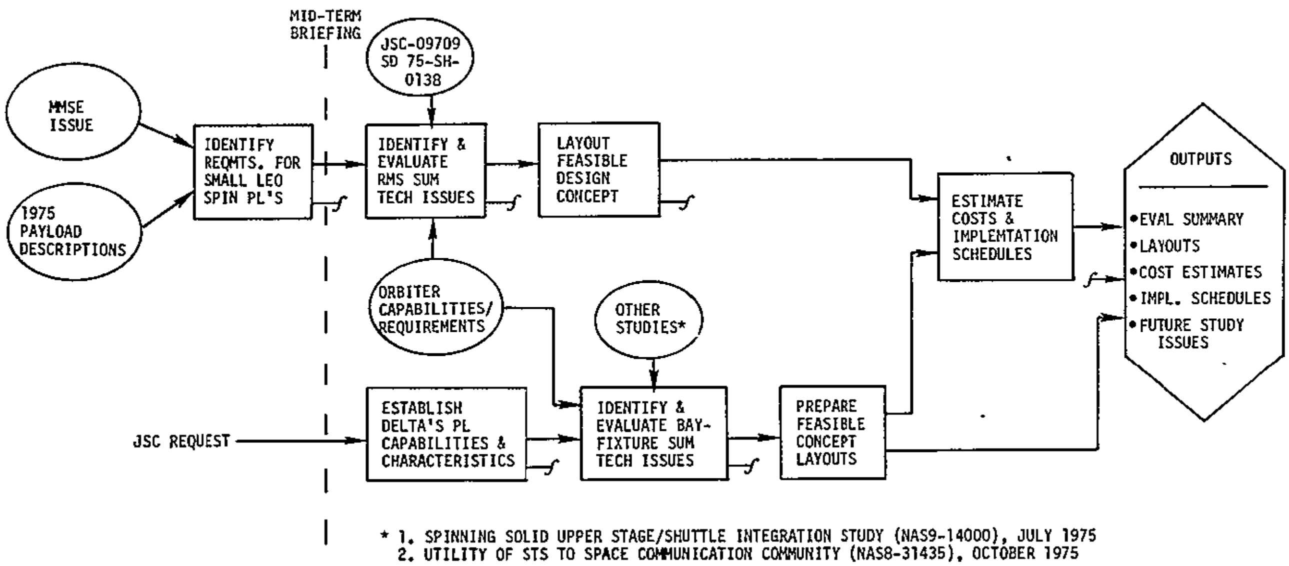


Figure 2-21. Subtask Logic Flow

The primary sources of data are indicated in Figure 2-21 (References 2, 3, 4, 5). In the case of the RMS, detailed design characteristics and capabilities are not yet available. Therefore, pertinent characteristics not directly defined by References 2 and 3 were derived, or interpreted, as necessary. Lack of specific design characteristics limits the depth/accuracy of the evaluation.

Results - Bay Platform Spin-up Mechanism

Requirements

The Reference 5 communications payload study states that 44 reimbursable launch spacecraft will be launched to geostationary orbits from 1979 to 1991. Table 2-8, based upon the July 1975 payload descriptions, Reference 6, indicates that as many as 59 high-altitude kick stage launches could fit into the Thor-Delta class. Reference 5 concluded that 70 percent of all geostationary launches and 50 percent of all commercial launches for the preceding 10 year

Table 2-8. Summary Thor/Delta Class and Low Orbit Class Payloads Requiring Spinup 1980-1991 Launches

Name	Designation	RPM	Weight Kg	Spin Provided by P/L/Tug/Sh	Size Diam/Length (in.)	Desired Alt (Km)/ Inclin (deg)	Pointing Accuracy Arc Sec	Stab (Arc Sec)/ Stab Rate (Arc Sec)	Flights	Tip Off Rate Max deg/sec	Reposition Capability	C.G. From Ref Plane (m)
Advanced Radio Astronomy Explorer	AS-05-A	-	600	-	1.83 x 2.47	35,786/(0, +5)	7200	1800/360	80/83/85/87*	0.1	Yes	1.16
Upper Atmosphere Explorer	AP-01-A	4	908.8	P/L	1.37 x 2.36	3510/(90, 70, 110)	1800	1800/180	84/89/90	TBD	N/A	TBD
Explorer - Medium Altitude	AP-02-A	4	271.8	P/L	1.37 x 1.83	37,038 (28.5, 0, 90)	1800	1800/180	81/82/84/88/90/91	TBD	N/A	TBD
High Altitude Explorer	AP-03-A	4	426.6	P/L	1.22 x 1.83	N/A (Elliptic plane)	1800	1800/180	83/85/86/87	TBD	N/A	TBD
Gravity & Relativity Satellite - LED	AP-04-A	0.2	650	P/L	2.2 x 3.6	500/937.5 (28.5, 90)	1800	1800/TBD	84/86/87	TBD	N/A	
Storm Satellite - B	EO-15-A	*	860	-	2 x 4	35,784 (0, 0.1)	N/A	N/A		-	Yes	1.0
Foreign Synchronous Met Sat.	EO-57-A	100	285.7	P/L	1.91 x 3.14	35,786 (0, 0.6)	1800	1800/360	81/82/84/86/88/90	1.0	Yes	1.44
Geosyn. Operational Envir. Sat.	EO-58-A	100	285.7	P/L	1.91 x 2.61	35,786 (0, 0.6)	1800	1800/360	81/82/83/85/87/88/ 89/91	1.0	Yes	1.44
Geosyn. Earth Resources Sat	EO-59-A	-	1475	-	4.47 x 5.26	35,786 (0, 0.1)	1800	1800/360	88/88/90/90	1.0	Yes	TBD
Gravity Gradiometer	OP-02-A	120	180	Sh	1 x 1	300 (90, 89.9, 90.1)	N/A	N/A	80*	N/A	N/A	1.37
Gravity Field Sat	OP-04-A	2	(3 at 91 ea.)	Sh	2 x 2.7	300 (90, 89.9, 90.1)	3600	3600/360	79*	1.0	N/A	1.1
Sphinx - B	ST-02-A	30	102	Tug	2.2 x 0.55	36,000 (20, 0, 40)	N/A	N/A		1.0 (Tug)	N/A	0.27
Sphinx - C	ST-03-A	1.0	216	Tug	2.2 x 0.76	36,000 (20, 0, 40)	N/A	N/A		N/A	N/A	0.38
International Telecom. Sat.	CN-51-A	-	1472.2	-	4.5 x 2.5	35,786/(0, 0, 0.1)	1800	1800/180	85/85/86/86/89/89/90/ 90/90/91/91	1.0	Yes	0.73
U.S. DOMSAT-A	CN-52-A	10	559	P/L	4.42 x 3.66	35,786/(0, 0, 0.1)	1800	1800/180	83/83/84/85/87/87/ 88/88/88/89/89/90/90	1.0	Yes	TBD
U.S. DOMSAT-B	CN-53-A	-	1472.2	-	4.5 x 2.5	35,786/(0, 0, 0.1)	1800	1800/360		1.0	Yes	0.73
Disaster Warning Satellite	CN-54-A	-	582.9	-	1.4 x 5.12	35,786 (0, 0, 0.6)	1800	1800/360	82/84/84/85/90	1.0	Yes	1.99
Traffic Management Satellite	CN-55-A	-	298.5	-	2.18 x 3.22	35,786 (2.15, 1.84, 2.46)	1800	1800/360	83/84/86/88/90	1.0	Yes	1.03
Foreign Communications Satellite	CN-56-A	-	307.9	-	2.2 x 2.36	35,786 (0, 0, 0.1)	1800	1800/360	82/83/84/90	1.0	Yes	0.99
Commun. R&D/Prototype	CN-59-A	-	1438	-	4.14 x 6.4	35,784/(0, 0, 0.1)	1800	1800/360	85/88/90	1.0	Yes	TBD
Foreign Communic. Satellite-B	CN-60-A	-	307.9	-	2.1 x 3.2	35,786/(0, 0, 0.1)	1800	1800/360	Included in CN-56	1.0	Yes	1.0
Lunar Halo Satellite	LU-03-A	60	1120	P/L	2.36 x 5.33	N/A	3600	3600/360	90	TBD		
Throw Away Detector		2	22.7- 90.9	Sh	0.3 x 0.76	Orbiter	TBD	TBD	2 or 3 flights in OFT (3 or 2 deploys per flt)	TBD	Unknown	Unknown
Amps Subsatellites		1-10	(2 at 275 ea.)	Sh	1.36 x 1.65	Orbiter	3600	360/360	2+ flights of 2 deploys each	TBD	Yes	Unknown
Environmental Perturbation Satellite	AP-05-A	-	1488	-	2.1 x 3.7	12,778/(55, 25, 85)	3600	3600/60	87/89	TBD	N/A	TBD
Gravity and Relativity Satellite	AP-06-A	-	349	-	2.6 x 2.1	N/A	1800	1800/3600	83/86/91	0.1	N/A	1.05
Earth Resources Survey Operational Satellite	EO-61-A	-	732.9	-	1.52 x 3.05	907.7 (99.098, 98.998, 99.198)	1800	1800/360	83/85/86/87/88/89/ 90/91	1.0	N/A	0.28
GEOPAUSE	OP-01-A	-	789.0	-	2 x 2.5	23,000 (90, 89.9, 90.1)	3600	3600/360	82*	1.0	N/A	1.37
Haley Comet Flyby	PL-19-A	-	580	-	3 x 3	N/A	TBD	TBD	85	1.0	N/A	TBD

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period will be launched with Thor-Delta configurations, and since the methods used have been demonstrated to be adequate and competitive, the same approach can be expected to continue unless the STS can competitively meet similar capabilities. Any estimate of the number of potential Thor-Delta launches must be only approximate since planning data is approximate and it is not known how users may trade off the final design of their spacecraft in order to accommodate Thor-Delta or Orbiter if both are available. For instance, the currently indicated spacecraft sizes/weights could be altered in many cases.

The maximum Thor-Delta fairing dimensions for 2914/3914 and 0914 Delta class payloads are shown in Figure 2-22. Comparison with Table 2-8 indicates a number of 1979-1991 payloads within or near Thor-Delta capability. For comparison, Table 2-9 lists the kinds of pre-STS launches which are designated for Thor-Delta or Scout launches from 1976-1980. Table 2-8 is primarily based upon the Reference 6 payload descriptions except where later data is available.

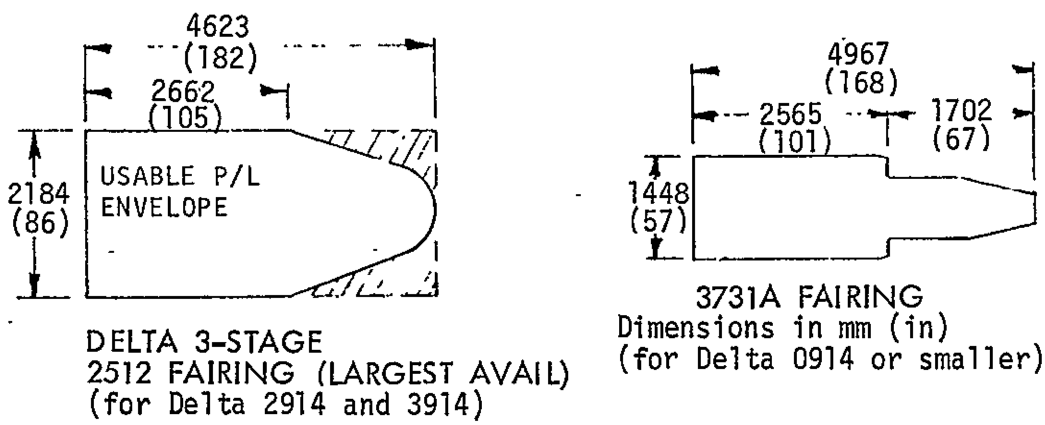

Figure 2-22. Two Sizes of Thor Delta Payload Fairings (Useable Envelopes)

In general, it was assumed that the goal would be to meet Thor-Delta capabilities with Orbiter. Reference 7 was used to establish pertinent capabilities which are summarized in Table 2-10.

Basic Design Approach Considerations

A number of design issues and options were identified and evaluated. Figure 2-23 summarizes these for the bay and RMS spin mechanisms. It was determined that an Orbiter attached spin-up approach is desired, as opposed to deploying a payload for a free-spin at a safe distance. The free-spin approach would require guidance and attitude control packages that would either be expendable or result in complex operations and recovery capability.

The Orbiter attached spin-up device should be erectable out of the bay in order to provide safe clearances for deployment. An attitude of about

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Table 2-9. Summary Thor/Delta Class and Low Orbit Class Payloads Requiring Spinning 1976-1980 Launches

Name	Designation	Geosynchronous P/L Placement Accuracy (3σ)						RPM	Weight Kg	Size in In. Diam/Length	Desired Alt (km)/ Inclin (deg)	Reposition Capability	C.G. From Ref Plane (m)
		Position Km			Velocity m/sec								
		Radial	Tangential	Normal	Radial	Tangential	Normal						
International UV Explorer	AS-21-A	46	2000	500	12	12	12	Typical Thor Delta	594	1.83/4.06	35,786/(28.9, 27, 22)	TBD	1.16
Radio Explorer (Synch Orbit)	AS-22-A	46	2200	550	12	12	22	Typical Thor Delta	580	1.8/2.5	35,786 (0, +5)	Yes	1.1
IUE Photometry	AS-24-A	46	2000	500	12	12	12	Typical Thor Delta	602	1.83/4.06	35,786 (28.9, 27, 32)	Yes	1.2
Synchronous Meteorological Satellite	EO-07-A	46	46	62	12.9	12.9	17.2	Typical Thor Delta	628	1.91/3.05	35,786 (0, -0.5, +0.5)	Yes	1.44
LANDSAT-C (ERTS-C)	EO-13-A	-	-	-	-	-	-	Typical Thor Delta	955	1.52/3.51	915 (99.1, 99, 99.2)	Yes	0.97
NIMBUS-G	EO-14-A	-	-	-	-	-	-	Typical Thor Delta	874.5	1.54/3.51	1090 (99.95, 99.9, 100)	Yes	0.97
Radiation Budget Satellite	EO-16-A	-	-	-	-	-	-	Typical Thor Delta	177	1/1	760 (50, 80, 80.5, 49.5, 79.5, 50.5)	No	-
EOS-A	EO-17-A	-	-	-	-	-	-	Typical Thor Delta	1171	2.13/4.88	740 (98.5, 98, 99)	N/A	TBD
SAGE	EO-18-A	-	-	-	-	-	-	Typical Thor Delta	127	0.97/1.2	611 (50)	No	TBD
Storm Satellite-A	EO-19-A	460	TBD	TBD	43	14	43	Typical Thor Delta	700	2/4	35,786 (0, +0.1, -0.1)	Yes	1
ITOS/NOAA	EO-63-A	-	-	-	-	-	-	Typical Thor Delta	337.2	1.22/1.5	1460 (101.74, 101.70, 101.78)	N/A	0.6
SEASAT-A	OP-09-A	-	-	-	-	-	-	Typical Thor Delta	1080	1.48/4.24	800 (108, 107, 109)	Yes	1.1
Orbiting Frog Otolith	LS-03-A	-	-	-	-	-	-	Typical Thor Delta	131	0.61/0.81	593 (TBD)	TBD	TBD
Communication Technology Satellite	CN-01-A	46	46	62	12.9	12.9	17.2	Typical Thor Delta	672	1.8/1.78	35,786 (0, -0.2, 0.2)	N/A	0.9
Lunar Polar Orbiter	LU-05-A	-	-	-	-	-	-	Typical Thor Delta	460	1.5/1.5	N/A	N/A	TBD

Table 2-10. Pertinent Thor-Delta Capabilities Summary

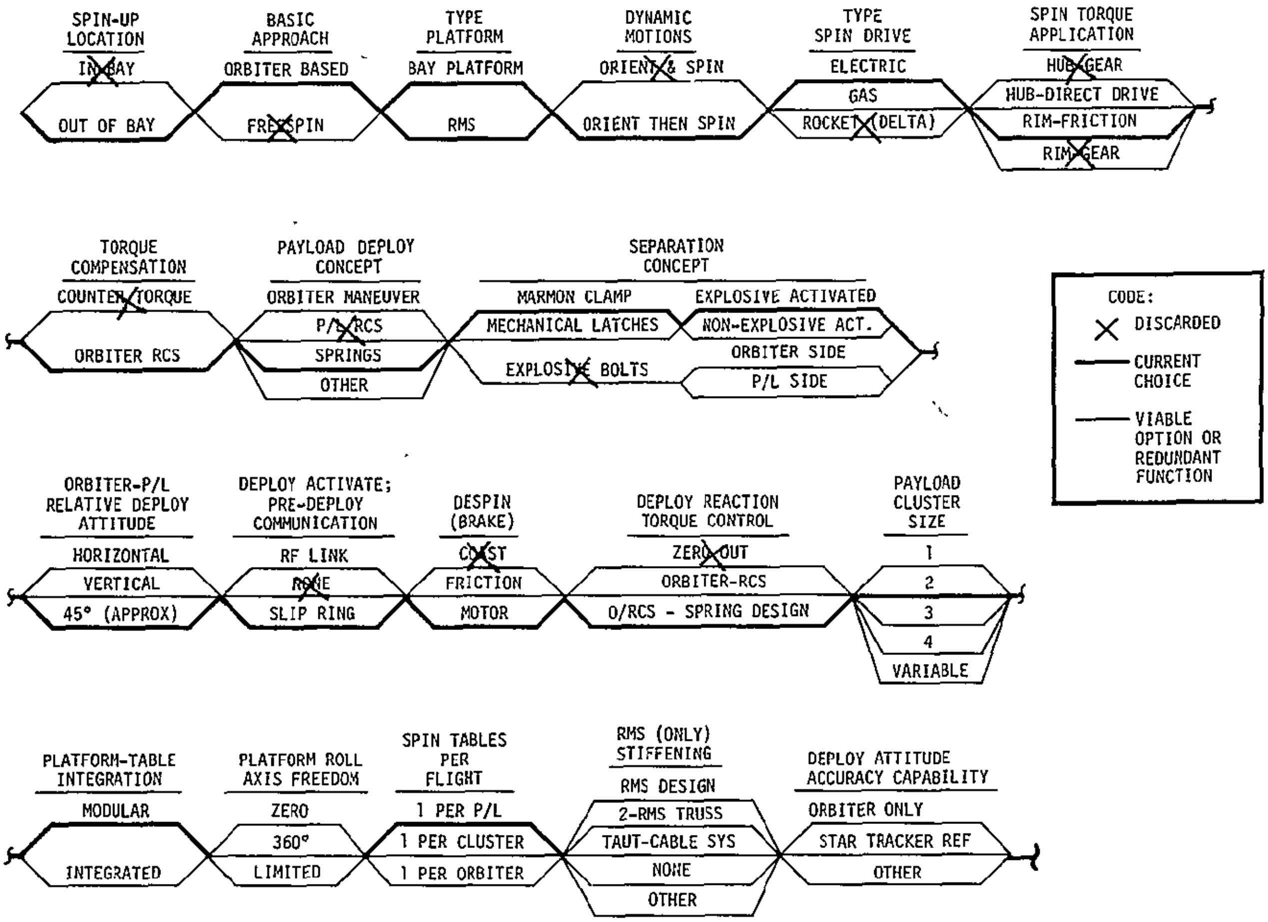
Parking orbit (nominal)	
Altitude	185 km (100 nmi)
Inclination	28.7°
Payload (for geosynchonous orbit, ETR launch	
Max size (including apogee motor)	See Figure 2-22.
Max weight (including apogee motor)	909 kg (2000 lb) (3914 Thor-Delta) 705 kg (1550 lb) (2914 Thor-Delta)
Static balance	± 1.3 mm (0.05 in) of rotation CL
Dynamic balance	± 0.02 radians between CL and a principal axis of inertia
Third stage deployment	
Attitude accuracy	± 0.5 arc-deg
Spin rates	4.2 to 15.7 r/s (40-150 rpm) ± 10%
Apogee motors	Thiokol TE-364-3 and 4
Weights (approx)	1136 kg (2500 lb) max (TE-364-4)
Standard stage integration hardware available	Spin table, interface sections, 3rd stage telemetry package.

45 arc-deg with respect to the Orbiter roll axis is preferred to maximize clearance of Orbiter at deployment. Figure 2-24 shows a layout to illustrate an erection mechanism to do this.

Using the useable payload volume lenth and diameter measurements for the largest Thor Delta fairing (Figure 2-22), it was found that two such payloads, with TE-364-4 apogee motors attached, could be clustered together. By clustering them one over the other, the Orbiter lateral c.g. is not impacted by the launch abort of only one payload, as a side by side arrangement does. This could allow landing with one payload in case it could not be deployed for some reason. Preliminary design indicates that each payload can be individually erected, providing maximum clearances from the other payload during spin-up and deployment. Some consideration was also given to using one spin table for multiple payload launches. However, the complication and time consumed during operations, as well as spin table latch-relatch complexities cause the practicality of such an approach to be very doubtful.

Deployment Dynamics

It was determined that a conventional marman-clamped, spring-driven deploy separation approach should be used. This is a proven, reliable, predictable method. When the payload is given a deployment delta-V a reaction



NOTE:
REDUNDANCY/RELIABILITY, FAIL-SAFE, COST, OTHER SUBOPTIONS EXIST FOR MOST OPTIONS LISTED.
MAXIMUM PRACTICAL STACKING DENSITY & PREVIOUSLY STATED GROUNDRULES & REQUIREMENTS ASSUMMED

Figure 2-23. Design-Issues Decision Path

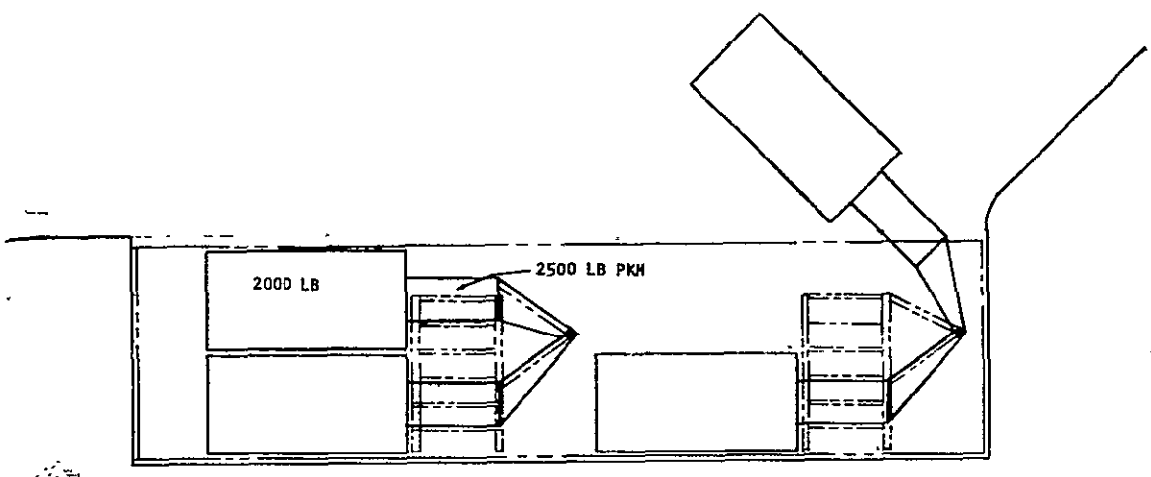


Figure 2-24. Payload Erection and Mechanism Concept

torque against the Orbiter inertia and RCS occurs. The impulse (force x time) applied to the payload must be controlled so that the Orbiter does not "dip" significantly to affect the deployment accuracy and tip-off rates.

The allowable force depends upon the moment arm with respect to Orbiter c.g., which could be about 4.6 m (15 ft) for rear-bay launches with an erection angle of 0.785 radian (45 arc-deg), or close to zero for forward-bay launches. The moment arm can be minimized by spin table location and erection angle.

The required force depends upon the desired delta-V to be given the payload and the deploy spring contact time over the impulse, which translates into spring stroke length. A practical spring length limit must be maintained for design feasibility. Reference 4 concluded that a 10,455 kg (23,000 lb) payload could be given a delta-V of 0.15 m/sec (0.5 ft/sec) with a stroke length of 0.5 m (1.65 ft) while maintaining a 7 to 1 Orbiter control authority. Reference 5 concluded that a delta-V of 0.61 to 1.22 m/sec (2 to 4 ft/sec) is desired. This is based upon minimizing the time to achieve a safe separation distance of 915 m (3000 ft) before the payload nutation angle builds up enough to significantly affect pointing accuracy at ignition. Since the largest Delta-class payload will not exceed 2270 kg (5000 lb), it appears that the required separation delta-V can be achieved. However, more in-depth analysis is warranted to trade off spring length and size, Orbiter control properties and erection table design and mounting as affecting spin table location and orientation.

Speed Control

A spin speed control feedback control concept was selected to simplify and optimize operations. (See Figure 2-25). The spin speed control concept is a first order servo that provides maximum spin torque until the set-in speed voltage reference is approached, as indicated by the tach-generator feedback voltage comparison. Motor drive is reduced, and continued (+) or (-) about zero to maintain the set-in speed within ± 5 percent. The motor is a reversing servo-type DC motor of sufficient size to achieve required angular momentum in a reasonable time. The servo electronics and/or computer software is straightforward and of minimum complexity. A vibration sensor detects excessive out of balance conditions and switches the servo to brake to zero speed as quickly as allowed, or as programmed. A backup mechanical brake may still be applicable but requires study.

As Figure 2-25 indicates, the spin speed mechanism is straightforward and is applicable to both the RMS and bay platform spin mechanisms. The hardware selections could differ considerably, however.

Pointing Control

Thor-Delta attains a pointing accuracy of .0085 radians (0.5 deg) at separation from the spin table. While Orbiter can maintain pointing to ± .0017 radians (0.1 deg), the absolute pointing uncertainty is up to 0.035 radian (2 deg) due to structural distortion between the navigation base and rear payload bay. Additional uncertainty can be caused by the spin platform

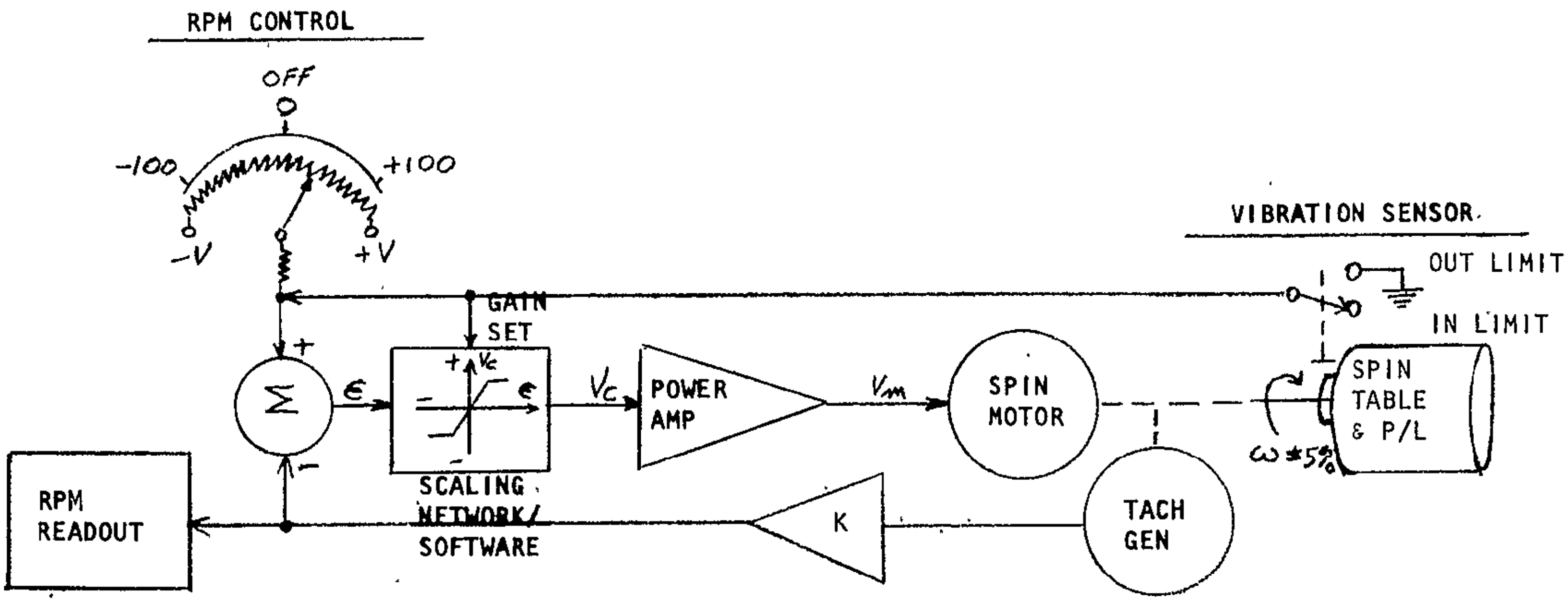


Figure 2-25. Spin Speed Control Concept

structure. Reference 5 concluded that the Orbiter induced error is acceptable since its rms contribution to the total errors, considering such as kick-stage thrust vector misalignment, is not a driving factor. An additional penalty to the payload spacecraft (2 to 3 percent of its weight) for orbit correction propulsion would result, however.

By adding a star tracker or other reference sensor at the spin table the errors could be reduced to that of the Orbiter RCS deadband capability. The Orbiter star tracker, or a modified version (reference the OST special emphasis study) could be applied. The cost of adding such an accuracy package, pro-rated over many launches, should be a small percentage of total costs. This concept should be studied further to make a decision before final design is firmed up since provisions for star viewing and/or auto-collimotor line of sights are necessary.

Other Issues

Other issues require further study before specifying design requirements. The communication requirements with the payload prior to deployment requires determination and the means of communication (if needed) decided. Cradle design needs more study. The payloads in Figure 2-24 have about 51 mm (2 inches) clearance inside the allowed top and bottom orbiter bay volume limits and about 102 mm (4 inches) clearance between payload envelopes. Additional support and/or a slight compromise on the payload envelope may be necessary for launch vibration clearances. This aspect has not been studied.

Spin Table Design

Figure 2-26 shows a spin table conceptual design which was developed for cost estimating and feasibility evaluation purposes. Rim drive electric motors were selected because of simple, reliable application for variable speed control and braking. The kick stage body and rocket nozzle protrudes into the center region in order to support the payload kick stage-spacecraft combination near the payload c.g. Two cradle supports (visible in Figure 2-26) take the loads off the spin table until erection for spin-up.

Payload Accommodation

It was found that at least four of the largest Thor-Delta class payloads could be flown on a single Orbiter flight without violating Orbiter c.g. requirements. Smaller payloads such as the Delta 0914 class could be clustered into groups or four for a possible total of at least eight. However, since it is not likely that many opportunities to launch so many payloads of this type at one time will occur, the accommodation of mixed payloads is perhaps more important.

The maximum single payload of the Delta class should not exceed 2045 kg (4500 lb) (909 kg [2000 lb] for spacecraft and 1136 kg [2500 lb] for a TE-364-4 kick stage). The maximum weight per cluster is then about 5448 kg (12,000 lb) allowing 1362 kg (3000 lb) for spin table, erection platform and support cradle weights. Spin table and cradle weights are based upon results from

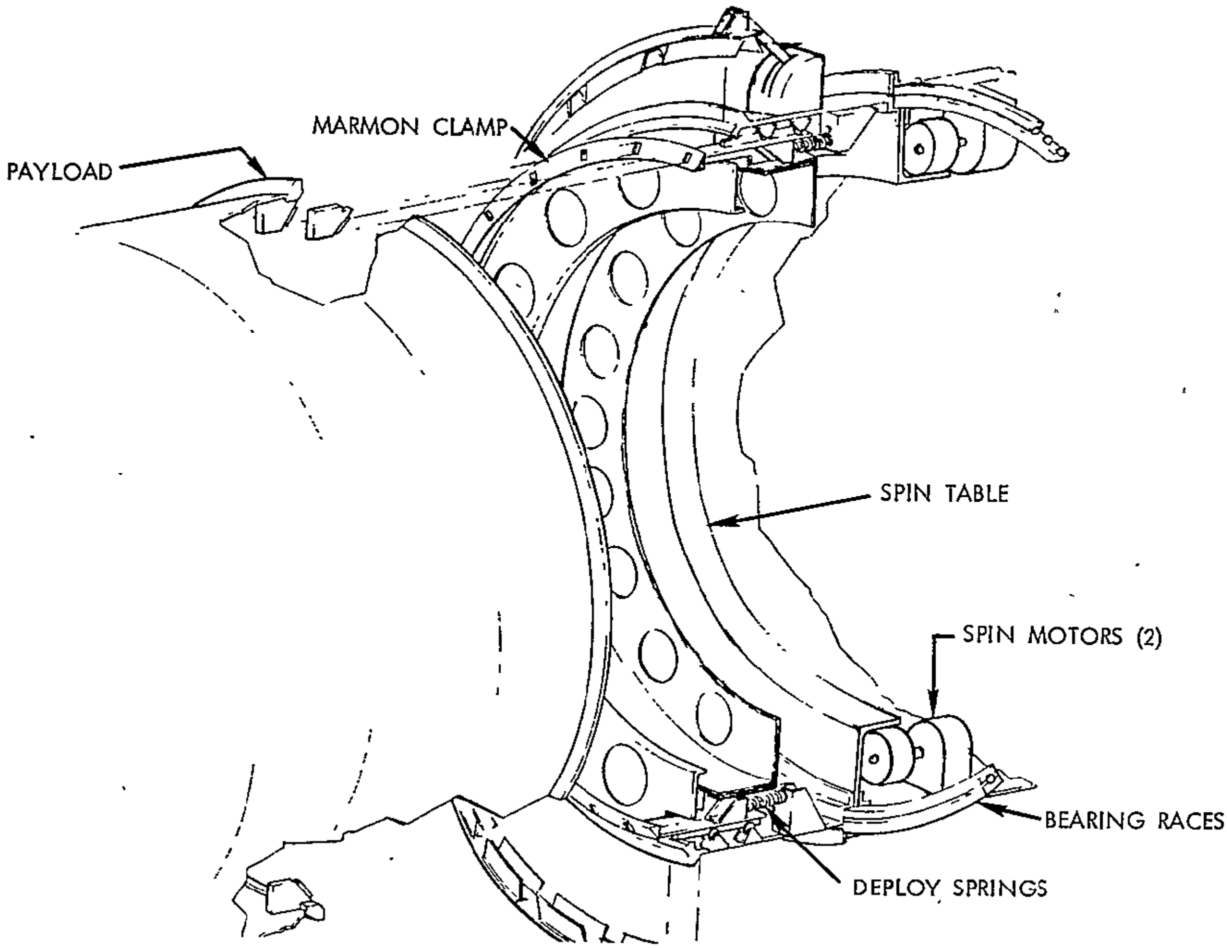


Figure 2-26. Spin and Deployment Mechanism

Reference 4. Total length of a cluster is about 7.9 m (26 ft). Therefore, about 10.4 m (34 ft) and 24,000 kg (53,000 lb) of payload capacity is potentially available to other payloads on single-cluster kick stage missions. A study to determine which kinds of other type payloads could potentially fly with kick stage payloads is recommended. Whether or not forward or rear-bay locations will be dominant may affect design.

Program Costs

Table 2-11 summarizes the estimated cost to design, qualify, and provide two sets of spin table, erection platform, and cradle hardware. Costs are extrapolated from the Reference 4 study which dealt primarily with a single payload concept. Extrapolation is based on relative complexity and materials factors for the two concepts. A four-cluster mechanism was costed as a measure of maximum costs. However, a two-cluster concept is recommended for initial procurement providing that a universal cluster approach is not practical. A universal approach would allow a basic mechanism to be configured with one to four modular spin tables attached, as applicable, with necessary add-on fixtures. By extrapolating between the concepts shown in Table 2-11,

Table 2-11. Cost Estimate Summary (Two Sum Sets) - $M

	SSUS/SI Study Est.		4-cluster mechanism	
	Plat./Table	Cradle	Plat./Table	Cradle
Engineering (design, analysis, test)	2.3	2.2	3.5	3.3
Manufacturing operations (including refurbishing qual test units)	2.1	2.0	6.3	6.0
Soft tooling	1.5	2.2	2.3	3.3
Quality program	0.5	0.4	0.5	0.4
Material	0.5	0.1	0.5	0.1
Installation, assembly and checkout	1.1	1.0	1.7	1.5
	$8 M	$7.9M	$14.8M	$14.6M

the two-cluster concept is estimated to cost $20M for two sets. The extrapolation assumes that complexity (and cost) is proportional to the cluster capability. Unique aft crew station control electronics are included in the estimate.

Program Implementation Planning

Figure 2-27 shows recommended time phasing for the major efforts needed to deliver two sets of Bay Platform Spin-up Mechanism hardware. It is based on having a space qualified capability ready for users when STS becomes operational. A flight test during OFT would be desirable, therefore a flyable set should be available in latter 1979.

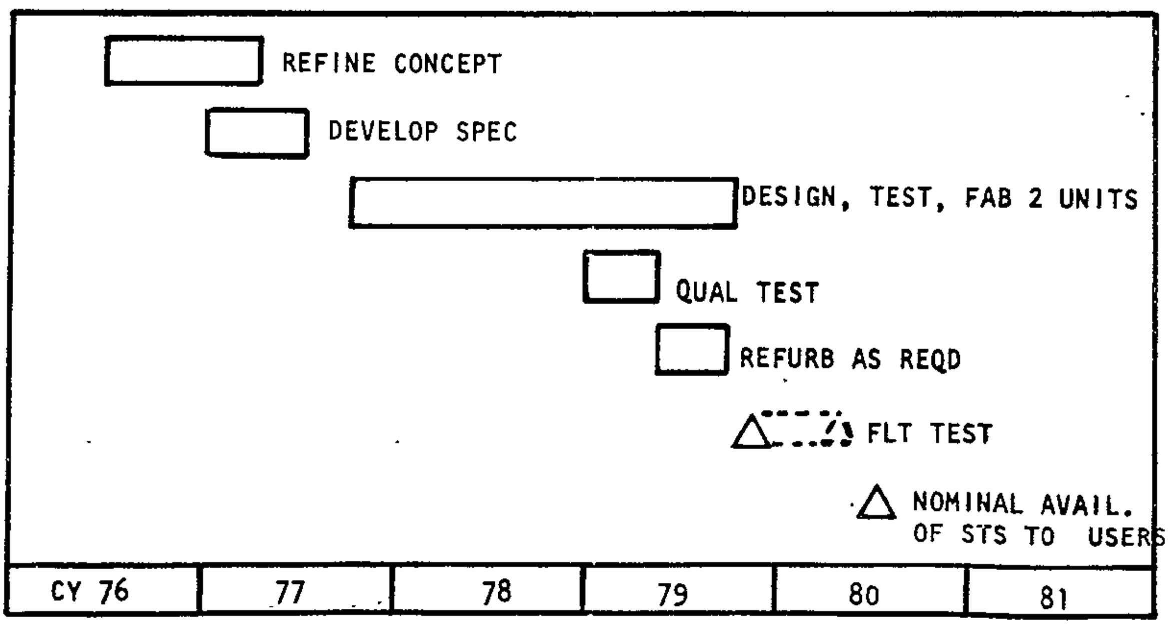


Figure 2-27. Bay Platform Spin Mechanism Program Implementation Plan

It is proposed that the ground qualification set be refurbished to flight worthiness since it is largely mechanical and modular.

Based upon a two-year design, manufacturing and test period, the purchase order needs to be let by latter 1977. A short preliminary design phase should precede the purchase order in order to thoroughly define all requirements and particular design studies. Procurement documents would be the primary output. As a result of the issues remaining, previously discussed, an in-depth concept study should begin as soon as possible. It would be preferred to complete the study before the preliminary design effort starts. However, it is felt that, realistically, funding could not be available before mid-1976. The early part of the study could define the preliminary design phase sufficiently to allow preliminary design to start, completing the concept study in parallel.

<u>Results - Remote Manipulator System Spin-up Mechanism</u>

Requirements

Low earth orbit spin stabilized satellites that could potentially be launched from the Orbiter with the RMS include the Throw-away Detectors (TAD's), the AMPS Spacelab subsatellites and others such as the Gravity Gradiometer and Gravity and Relativity spacecraft. Spin rate requirements range from 0.021 to 12.6 radians per second (0.2 to 120 rpm). Weights range from 22.8 to 91 kg (50 to 200 lb) for the TAD's to 600 kg (1320 lb) for the AMPS subsatellites, which are currently envisioned as Atmospheric Explorer type spacecraft, although analysis is still in process. The RMS is currently envisioned for TAD's deployment by the TAD agency. No data is available on currently planned means to deploy other satellites. TAD and AMPS plan satellite recovery with the RMS, however.

The TAD and AMPS employ two to three launches per flight with two or more flights planned. The gravity/magnetic field measurement satellites apparently will also involve up to three launches per flight. The Atmospheric Explorer (AMPS subsatellites) and presumably others, have complete attitude control capabilities. This is important due to the limited absolute accuracy capability of RMS pointing, given that an accuracy package is not provided.

RMS-Orbiter Capability

The capability and characteristics of the RMS are major factors in the technical feasibility of an end effector spin-up mechanism. Table 2-12 summarizes the required RMS characteristics as pertinent to this study. References 2 and 3 were primary sources of the listed data. RMS design data is not yet available.

Basic Design Approach Considerations

Due to the range of spin speeds potentially required, it was concluded that an electric motor driven spin-up is the simplest re-useable approach. It was also concluded that the likely preferred mode of operation is to extend the RMS straight out and "lock" it. The Orbiter would be relied upon

Table 2-12. RMS Characteristics

. Deployment of payload	Up to 5 per mission
. Payload dimensional envelope accommodated	4.6 m (15 ft) diam x 18.3 m (60 ft) long
. Payload size accommodated (deployment)	29,545 Kg (65,000 lb) (max)
. Clearance envelope required for payload	± 76.2 mm (3 in)
. End effector changeout	On orbit capability
. End effector speed (without payload attached)	0.61 m/sec (2.0 ft/sec) max (Nom)
. End effector speed (with 32,000 lb payload)	0.061 m/sec (0.2 ft/sec) max.
. Payload to end effector interface alignment in roll or yaw;	± 2.6 m radians (0.15 deg)
in X, Y, and Z	2.54 mm (0.1 in)
. End effector with grapple fixture misalignment accommodations requirement in roll,	± 0.26 rad (15° deg)
pitch or yaw; in Y and/or Z	101.6 mm (4 in)
. Wrist roll rate	.11 rad/sec (6.3 deg/sec) max (nom)
. Stopping distance with 32,000 lb payload	0.61 m (2 ft)
. Time to deploy payload	7 min. max
. Closing rate of payload at contact with payload retention mechanism, or handling aids	0.0305 m/sec (0.1 ft/sec) (max)
. Tip stiffness	0.57 mm/N (0.1 in/lb)
. Max. force perpendicular to tip	66.7 N (15 lb)
. Payload stabilization	
Attitude rate	± 1.75 mrad/sec (0.1 deg/sec)
Attitude hold	± 76.2 mm (3 in)
. Attitude error	± 0.26 rad ± 0.0175 rad (15 deg ± 1 deg) assuming 0.175 mrad/sec (0.01 deg/sec) orbiter stability
. Linear tipoff motion	0.061 m/sec (0.2 ft/sec)
. Angular tipoff motion	
LDEF	0.53 mrad/sec (0.03 deg/sec) max
All other payloads	1.75 mrad/sec (0.1 deg/sec) max
. Required orbiter/end effector stationkeeping characteristics during payload release ops	
Relative velocity	0 ± 0.0305 m/sec (0.1 ft/sec)
Translation range	± 1.524 m (5 ft)
. Joint stiffness	
Shoulder yaw	1.6×10^6 N-m/rad (1.40×10^7 in-lb/rad)
Should pitch	1.6×10^6 N-m/rad (1.40×10^7 in-lb/rad)
Elbow pitch	8.5×10^5 N-m/rad (7.53×10^6 in-lb/rad)
Wrist pitch	1.6×10^5 N-m/rad (1.43×10^6 in-lb/rad)
Wrist yaw	1.6×10^5 N-m/rad (1.43×10^6 in-lb/rad)
Wrist roll	1.6×10^5 N-m/rad (1.43×10^6 in-lb/rad)
. Arm Inertia	
Upper arm	2.59×10^{-5} m^4 (62.33 in^4)
Lower arm	1.19×10^{-5} m^4 (28.55 in^4)

to provide pointing and stability. The RMS could be oriented to minimize the torque couple about Orbiter c.g. when the payload is deployed, if necessary.

It appears possible to place an accuracy package on the spin-up mechanism to achieve accurate absolute pointing accuracy and stability. A star tracker and/or inertial unit appears to be the most direct means of doing this, although an autocollimotor concept may be possible. The Orbiter and/or RMS would then be controlled with respect to the attitude reference provided by the accuracy package. However, it was concluded that such provision may not be needed as long as use is limited to relatively small low orbit (non-kick stage) satellites. When the RMS dynamic characteristics become more accessible, the technical feasibility of high accuracy pointing should be studied. With accurate control it may be feasible to also deploy some spinning kick stage payloads by this method.

Evaluations

The RMS was considered to be a uniform long slender rod for this preliminary evaluation. While the characteristics of this equivalent rod may be difficult to judge due to the number of joints and mechanisms and undefined structure, nonetheless some illumination of the basic dynamic properties can be judged with such simplification.

Figure 2-28 illustrates the model evaluated in order to establish the RMS tip deflection and natural frequency characteristics. Gyroscopic effects will tend to constrain the payload end of the rod fixed in attitude. Ignoring

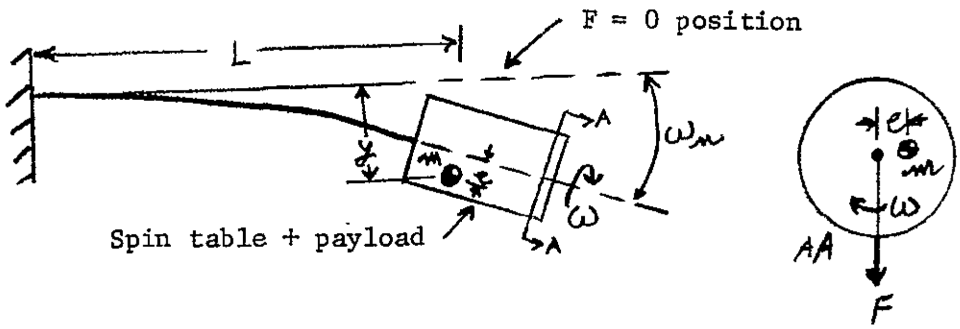


Figure 2-28. Model of RMS, Spin Table and Payload Dynamics

gyroscopic and resonance effects initially, the tip displacement force is:

$$F = ma_n = me\omega^2 ,$$

Calculation of the force for $e = 1.3$ mm (0.05 in) results in the values tabulated in Table 2-13 for various payload weights and spin speeds.

Table 2-13. Estimated Tip Deflection Force - gm (lb)

P/L Weight Kg (lb)	Radian/Sec (rpm)		
	.1(1)	1 (10)	10 (100)
22.7 (50)	.03 (0.00007)	3 (0.007)	300 (0.7)
22.7 (500)	.3 (0.0007)	30 (0.07)	3000 (7)
22.7 (5000)	.3 (0.007)	300 (0.7)	30,000 (70)
Mass unbalance = 1.27 mm (0.05 in)			

The characteristics of the payload mass and the spring constant of the "long rad" result in harmonic vibration where

$$\omega_n = \sqrt{\frac{K}{m}} = \sqrt{\frac{3\ EI}{L^3 m}}$$

with the system driven by $F = m\ e\ \omega^2$, the tip displacement at the driven angular rate, ω, is related to the natural frequency, ω_n, by:

$$Y_{max} = \frac{F}{k}\left[\frac{1}{1-(\frac{\omega}{\omega_n})^2}\right] = y_{static}\left[\frac{1}{1-(\frac{\omega}{\omega_n})^2}\right]$$

when $\omega = \omega_n$ (resonance), y_{max} can go to infinity if there is no damping in the system. Fortunately all systems have dampening, however, information on the RMS damping is not available. Figure 2-29 illustrates how the ratio,

$$\frac{y_{max}}{y_{static}} = \text{amplification factor} = \frac{1}{1-(\frac{\omega}{\omega_n})^2}$$

is affected as a function of $\frac{\omega}{\omega_n}$. It can be seen that attenuation begins quickly above resonance. Also, from the above relationships, y_{max}/y_{static} rapidly approaches 1 for ω less than ω_n. Therefore, the point of resonance is the concern. If the damping factor is 0.2 to 0.5 or larger, little or no amplification occurs even at reasonance. As Table 2-13 indicates, tip deflection will be small, except at the highest payload and weight regions, even with some amplification. Tip deflection is specified as

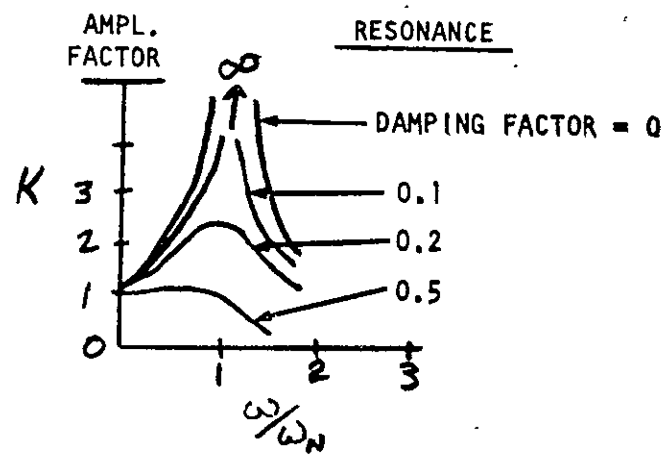


Figure 2-29. Amplification Factor Characteristics

less than 0.057 mm/N (0.1 in/lb). On this basis it appears that only 10 radian/sec (100 rpm) class payloads should pose a problem.

However, further analysis with a detailed characteristics model of the RMS is necessary to determine operating limits. In addition to the RMS construction, the payload gyroscopic phenomena affects the resonant frequency by constraining the payload end as a function of spin speed. The resonant frequency and dampening characteristics must therefore be carefully evaluated.

Design Concept

A design concept illustrated in Figure 2-30 was developed for feasibility and cost evaluations. It consists of a standard end-effector-to-RMS interface (not yet designed), a motor-tachometer assembly, and a modified end-effector interface suited for payload deployment. Delta-V springs and grapple capability is assumed. The standard grapple fixture (to be developed) may be satisfactory. The standard end-effector interfaces must be capable of making electrical interfaces.

The DC torquer type servo motor and tachometer are off-the-shelf designs (Inland T-6205 motor and TG-S113 tachometer). The motor produces a peak torque output of 13.6 N-m (10 ft-lb). The resulting spin-up capability is shown in Figure 2-31. The speed control concept is similar to Figure 2-25, previously discussed.

The payload diameter is based upon a homogenous cylinder. As can be seen, low spin rate, low weight payloads can be brought up to speed in a reasonable time and would probably be adequate for most potential payloads considered. However, a larger motor may be desired. In addition, a redundant braking device may be desired to brake a spun up payload in case it is not deployed for some reason and/or if the primary motor fails. These are issues needing further study.

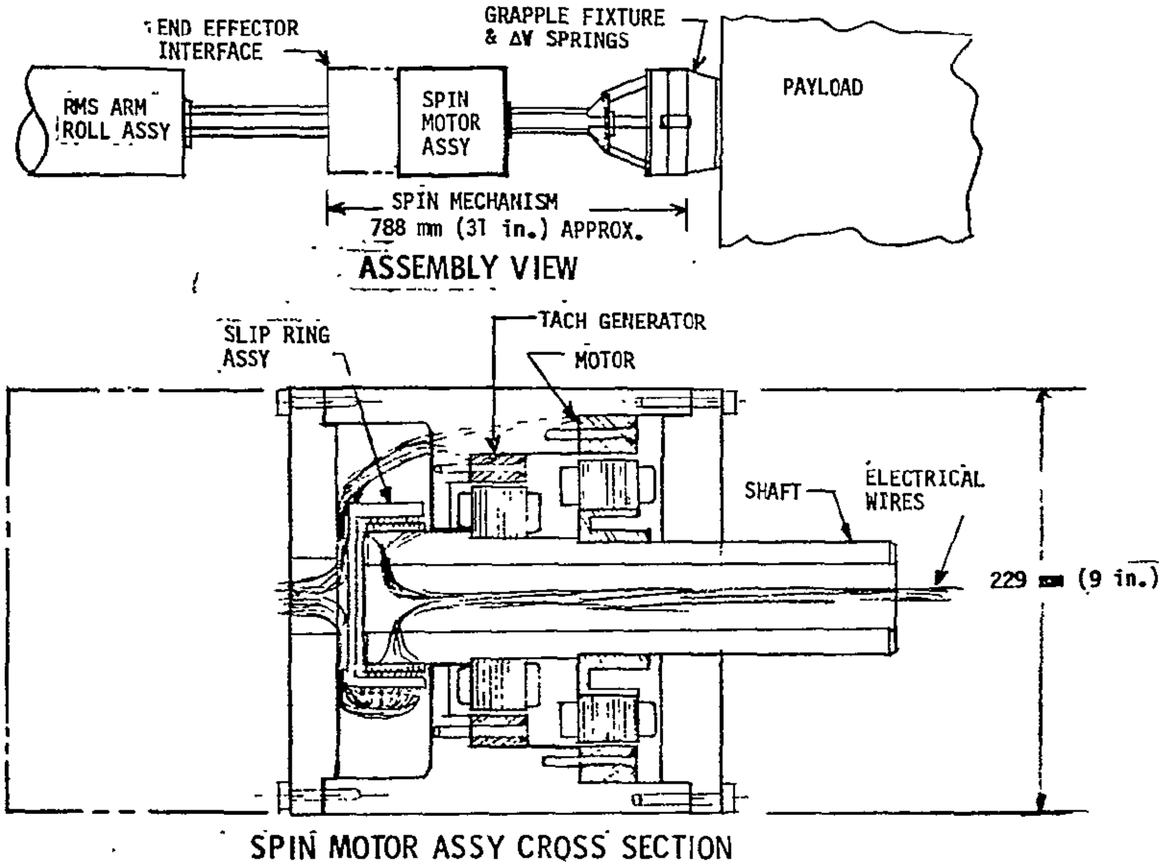


Figure 2-30. RMS Spin-up Mechanism Design Concept

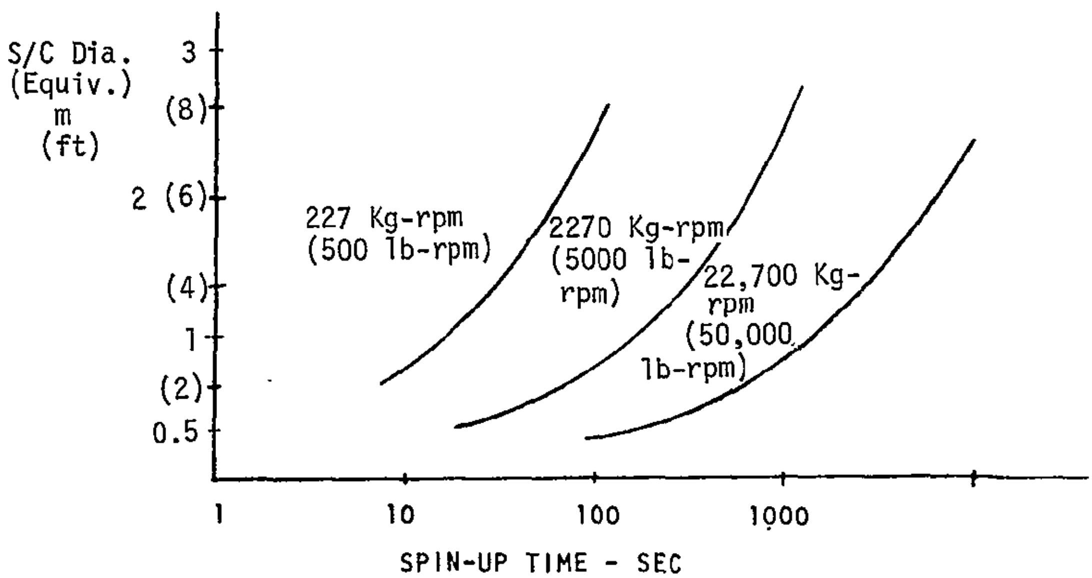


Figure 2-31. Spin-up Capability of Conceptual Design

Program Costs

Table 2-14 summarizes ROM cost estimates made for the RMS spin-up mechanism. Cost relationships for Orbiter type hardware based on weight and complexity factors were utilized. The cost shown provides two flight units, including unique control electronics.

Table 2-14. Cost Estimate Summary

	$M
Engineering (design, analysis, test)	0.43
Manufacturing operations	0.39
Soft tooling	0.28
Quality program	0.09
Material	0.09
Installation, assembly and checkout	0.22
	$ 1.5 M

Program Implementation Planning

Figure 2-32 illustrates the basic efforts and time phasing to bring the RMS spin-up mechanism into the flight inventory. The TAD's are planned for deployment during OFT. Delivery of a flight unit is required prior to mid-1979 to permit integrated ground testing. Meanwhile, the qual-test unit can be undergoing refurbishment to flight worthiness. Since the mechanism

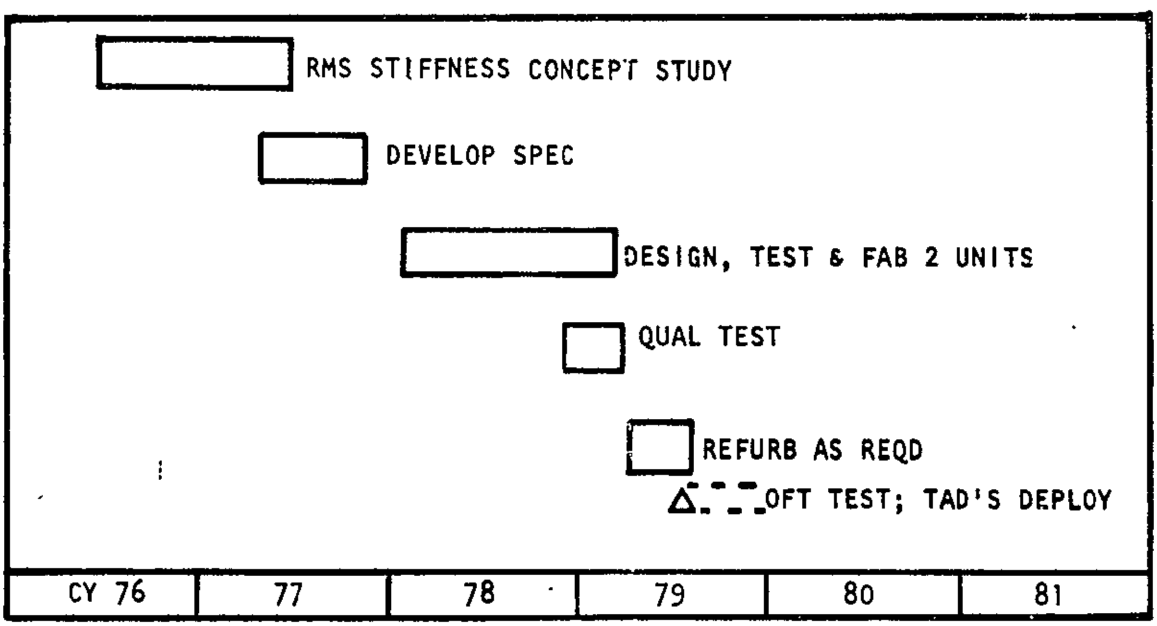


Figure 2-32. Program Implementation Schedules

itself is basically simple and straight forward, a design period of 15 to 18 months should be adequate. The purchase order should therefore be let by early 1978. A prior preliminary design effort is needed to define requirements and develop procurement documents. However, it is recommended that an early dynamics analysis be performed to insure technical feasibility.

Conclusions

Preliminary evaluations indicate that both the bay platform and RMS spin-up mechanism are technically feasible. The capability limits of the RMS spin mechanism is questionable, while the ability of the bay platform spin mechanism to handle all Delta-class payloads appears assured. A thorough dynamics analysis of the combined RMS-spin mechanism system is needed before the RMS spin-up limitation is understood. Also, the standard RMS end-effector design affects the spin mechanism feasibility. Electrical signal interfacing must be possible over the RMS-end-effector interface.

Key issues to be addressed for the bay platform spin mechanism includes:

1. Need for a pointing accuracy package
2. Payload cradle support/clearance envelope
3. Mixed payload potential
4. Optimum cradle and erection mechanism design approach.

Recommendations

It is recommended that in-depth concept feasibility studies be started as soon as funding will allow.

PAYLOAD/ORBITER ELECTRICAL CABLES

Introduction

The current overall program plan is for all payloads to be responsible for the design and procurement of the electrical cable sets required to connect to the Orbiter-supplied payload electrical panels. (An exception is the recent direction for Rockwell to supply those for Spacelab configurations.) Since a particular payload may be located in different positions in the bay from mission to mission, the cost of cabling could be significant for any one payload. When all payloads are considered, the total cost to the space program is expected to be large anough to warrant consideration of commonality of cables between different payload configurations and between flights. Such common cables could then be identified as MMSE.

Objective

The objective of this study was to establish a preliminary concept for the size, functional capabilities, and lengths of a common set(s) of electrical cables which will satisfy the requirements for each known payload configuration.

Approach

To accomplish the above objective the study was divided into three basic tasks. (1) Define the electrical requirements including functions, payloads and orbiter interfaces, EMI classification and wire types for all payload types, (2) identify cable configuration options considering wire tray configuration, cable composition, wire stowage limitations and the station location of the payload interface, (3) evaluate the options utilizing the criteria of degree of commonality, weight, electrical losses, installation complexity, reliability implications, and cost.

Results

Task 1 - Requirements Definition

Five general types of payloads were investigated during this study.

1. Spacelabs (sortie)
2. Automated
3. NASA with IUS
4. DOD with IUS
5. DOD free flyers

Twenty different configurations of these five types were identified along with their respective electrical interface station locations (Table 2-15). The LCMS, specifically the EOS, was chosen as representative of the automated type of payload, as it was the only payload of this type for which the electrical requirements were defined. The identified Spacelabs interface locations were taken from an ERNO "Utility Interface Considerations" briefing given to Washington in March 1975. Those for the NASA and DOD payloads with IUS were obtained from solid propellant IUS interface definition study documentation (Reference 8).

Table 2-15. Payloads Electrical Interface Location

Payload	Interface Station
Spacelab*	
3 pallet	694
5 pallet	690
Small module + 2 pallet (SM + 2P)	748
Small module + 3 pallet (SM + 3P)	748
Large module + 1 pallet (LM + 1P)	756
Large module + 2 pallet (LM + 2P)	756
Module	815
Automated	
LCMS (representative)	1069
NASA/IUS	833, 951, 1010, 1128
DOD/IUS	
RSA, GPS or DSP with 2-stage IUS	1010
RSA, GPS or DSP with 2-stage IUS (2 units)	833 (1st) 1128 (2nd)
RSA, GPS or DSP with 3-stage	951
DOD Free Flyers	
DMSD	710
SOSS	1069

*Interface is at forward edge of forward end cone flange for configurations with modules and at centerline of igloo for pallet only configurations.

A recent Shuttle program effort defined the Orbiter-to-payload electrical interface panel configuration and the estimated maximum interface wire requirements for the five payload types mentioned above. Table 2-16 lists the wire requirements by type and EMI class for the various functions for each of the payloads. The wire function will affect the final configuration of the wire bundles within a cable set but is assumed to have no effect on the concept definition detail to be developed in this study. Therefore, wire function will be given no further consideration. Table 2-17 summarizes the payload electrical requirements showing the orbiter interface station and the number of the various types and classes of wire.

The Orbiter interface panels at station 576 are being designed with a total capability consistent with the data generated in the aforementioned Orbiter-to-payload study effort. This capability can affect the final selection of the configuration of the MMSE cable set(s) and is shown in Table 2-18. Figure 2-33 depicts the current configuration of the interface panel at station 576.

Table 2-16. Payload Wire Requirements

Item	Thru Patch Distr.	Function	Spacelab Pallet-Only	Spacelab Module + Pallet and Module-Only	NASA/IUS	DOD/IUS	LCMS	EMI/EMC Classifica.
1	PS	NASA TM & CMD			6 TSP		6 TSP	RF
2	PS	DOD			10 TSP		10 TSP	RF
3	PS	PCMMU	2 TSP	2 TSP				RF
4	PS	PSS/DOD			10 TSP	10 TSP		RF
5	PS	DOD AUDIO						RF
6	PS	DOD INTER-COM						ML
7	PS	COMSEC 1&2						RF
8		ORBITER C&W	5 TSP	5 TSP	5 TSP	5 TSP	5 TSP	ML
9	MS	ORBITER EMERGENCY	4 TSP	4 TSP			4 TSP	ML
10	PS	EMERGENCY SAFING (C345)						ML
11	PS	C&W TONE	1 TSP	1 TSP				ML
12		MASTER ALARM LIGHT	1 TSP	1 TSP				ML
13	MS	P/L C&W INPUT	10TS5C 10TSP	10TS5C 10TSP	10TS5C 10TSP	10TS5C 10TSP	10TS5C 10TSP	ML
14	MS	P/L SAFING CMD	10TS5C	10TS5C	10TS5C	10TS5C	10TS5C	HO
15	MS	SYSTEM MONITOR	12TS5C 15TSP	12TS5C 15TSP				ML
16	MS	SYSTEM ACTIV.	16TS5C	8TS5C				ML
17		GN&C UPDATE	4TSP	4TSP				RF
18	PS	GN&C P/L PTG	8 TSP					RF
19	MS/PS	TIMING 1.152 MHz						RF
20	PS	TIMING	5TSP 1CX	5TSP 1CX	3TSP	3TSP	3TSP	RF
21	PS	RECORDING	30TSP		2TSP	2TSP		ML
22	PS	AUDIO		12TSP 2TP				TSP = ML 1 TP = HO 1 TP = ML
23	PS	TV	5 CX	5CX				RF
24	PS/MS	S-BAND-FM			1TSP 2CX	1TSP		RF
25	PS	KU-BAND	6CX	6CX			4CX	RF
26	MS/PS	MISSION-UNIQUE	88TSP 62TP 6CX	43TSP 15TP	44TSP 25TP 3CX	44TSP 25TP 3CX	145TSP 87TP 6CX	77 TSP = ML 9 TSP = RF 188 = HO
27	MS/PS	DOD P/L C&W						RF
28		C&W SERIAL I/O	4TSP					RF
29	PS	PSS/DOD						
Total			173TSP 62TP 48TS5C 18CX	102TSP 17TP 40TS5C 12CX	91TSP 25TP 20TS5C 5CX	75TSP 25TP 20TS5C 3CX	183TSP 87TP 20TS5C 10CX	

Table 2-17. Payload Electrical Requirements Summary

PAYLOAD TYPE	POWER ORB INTF	DATA AND CONTROL ORB INTF	ML TP	ML TSP	ML TS5C	HO TP	HO TS5C	RF TSP	RF CX	TOTAL TP	TOTAL TSP	TOTAL TS5C	TOTAL CX	TOTAL WIRE
SPACELAB														
PALLET	695	576	0	145	38	62	10	28	18	62	173	48	18	728
MODULE + PALLET	695	576	1	87	30	16	10	15	12	17	102	40	12	450
MODULE	695	576	1	87	30	16	10	15	12	17	102	40	12	450
DoD/IUS	N/A	576	0	57	10	25	10	18	3	25	75	20	3	303
NASA/IUS	N/A	576	0	57	10	25	10	34	5	25	91	20	5	337
DoD-FREE FLYER	695	576	← TBD →											
OTHER														
LCMS	695	576	0	149	10	87	10	34	10	87	183	20	10	650

Table 2-18. Payload 576 Service Panel Feedthroughs-Functional Assignment

A7J1 RF 44 TSP (88)
FM S-band - 1 TSP
PS DOD - 10 TSP
GN&C Update - 4 TSP
PDI - 10 TSP
C&W Serial I/O - 4 TSP
Mission Unique - 9 TSP
NASA TM & CMD - 6 TSP

A7J2 RF 45 TSP (90)
PCM M/U - 2 TSP
GN&C P/L Pointing - 8 TSP
Timing - 5 TSP
P/L RCDR - 30 TSP

A7J3 HO 29TP, 5TS5C (88)
Mission Unique - 29TP
P/L Safing CMD - 5TS5C

A7J4 HO 30TP (60)
Mission Unique - 30TP

A7J5 ML 26 TSP (78)
Mission Unique - 26 TSP

A7J6 ML 18 TSP, 1 TP (56)
Orbiter Safing - 5 TSP
C&W Tone - 1 TSP
Audio (Accu) - 12 TSP
Audio (Accu) - 1TP

A7J7 ML 26 TSP (78)
Mission Unique - 26 TSP

A7J8 ML 13TS5C, 5 TSP (93)
System Activation - 8TS5C
P/L C&W Input - 5TS5C
Orbiter Safing - 5 TSP

A5J1 HO 30TP (60)
Mission Unique - 29 TP
Audio (Accu) - 1TP

A5J2 - Spare

A5J3 ML 10 TSP, 11TS5C (96)
System Monitor - 6TS5C
P/L C&W Input - 5TS5C
P/L C&W Input - 10 TSP

A5J4 HO 5RS5C (30)
Payload Safing CMD - 5TS5C

A5J5 ML 27 TSP (81)
Mission Unique - 27 TSP

A5J6 ML 14TS5C (84)
System Activation - 8TS5C
System Monitor - 6TS5C

A5J7 ML 25 TSP (75)
Orbiter Emergency - 4 TSP
Orbiter C&W - 5 TSP
System Monitor - 15 TSP
Master Alarm Lt. - 1 TSP

A5J8 - Spare

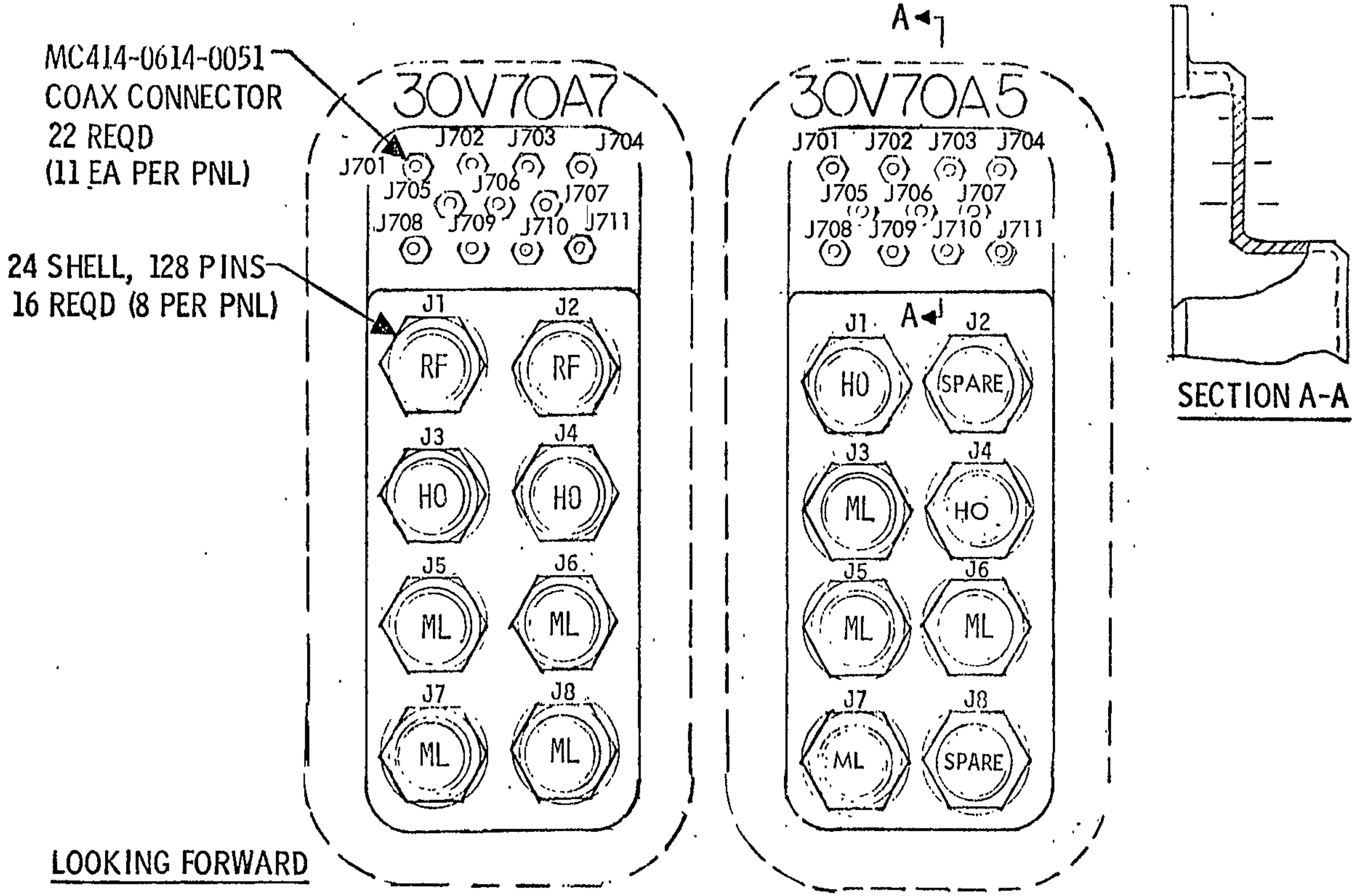


Figure 2-33. Feed-through Connector Panel Port Side Bulkhead $X_o 576$

The payload electrical interface stations are summarized in Figure 2-34 along with the Orbiter/payload electrical panel locations and capability. This figure and Table 2-17 present the requirements data necessary to define the cable configuration options and to evaluate their relative merits. Note the four additional Spacelab pallet interface locations (1 pallet-length apart). They have been added for consideration because the various combinations (as yet undefined) of pallets and automated payloads could result in these pallet locations.

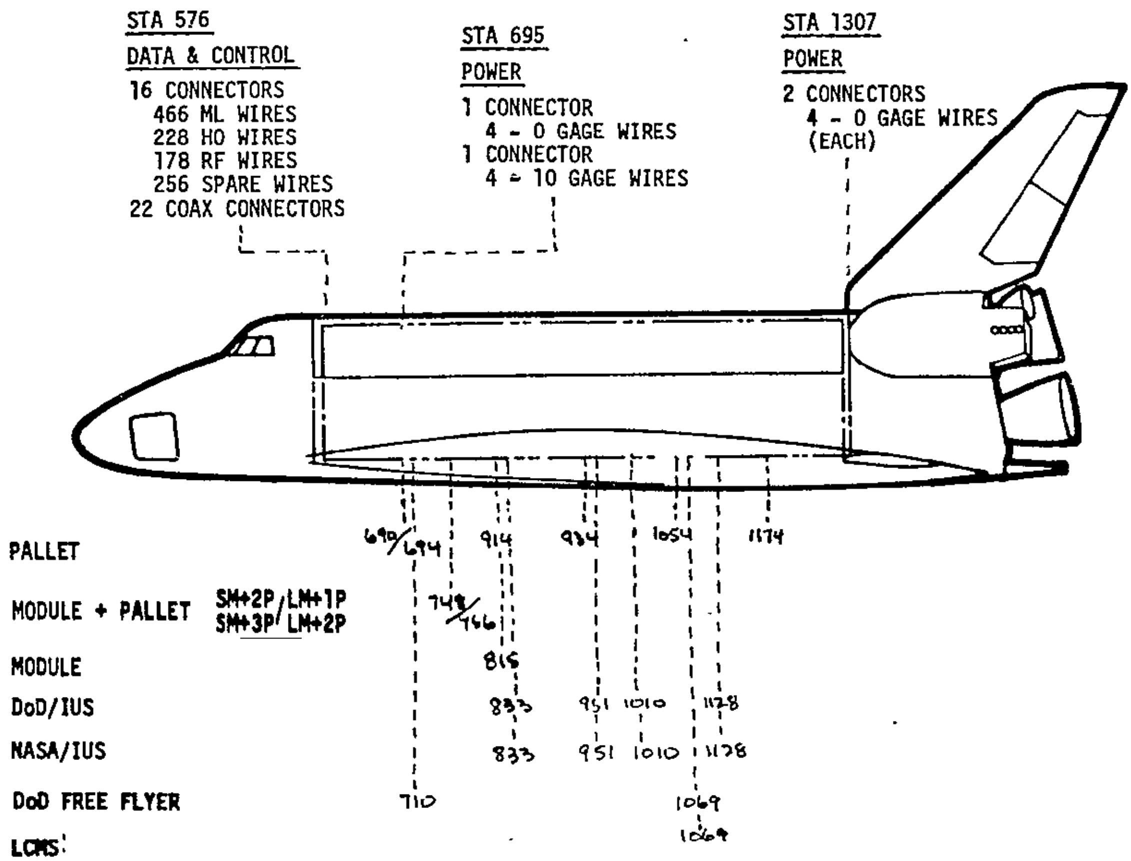


Figure 2-34. Orbiter and Payload Interface Locations

Task 2 - Options Identification

Five options have been identified which could satisfy the payload electrical requirements at the identified payload locations. They are:

1. A separate electrical cable set for each individual payload This is the current method planned for use by the payload disciplines.

2A. Separate individual cable sets designed to provide multiple payload interfaces within reasonable close proximity; with the total complement of cables providing support to the identified payload locations.

2B. Same as 2A, but with the capability to provide coverage of the entire bay with the proper selection of the number and length of the cables.

3. One cable set long enough to provide support to the aft areas of the payload bay; and then be looped and stowed as necessary to satisfy requirements any place within the payload bay.

4. One cable set permanently installed in the midbody wire tray, with outlet receptacles located strategically along the sidewall. This concept would require an additional cable set to connect from the sidewall outlet to the payload interface. As with Option 2B, proper selection of the number of outlets and the length of the companion short cable set can provide support to payloads at any location within the bay.

5. One cable set long enough to provide support to the forward section of the payload bay, and capable of having additional sections of identical sets connected to provide extended support to the remainder of the payload bay.

Option 3 was dropped from further consideration because of the excess weight and stowage problems created when a cable with this configuration would be used by payloads located forward in the bay. Figure 2-31 indicates the full length of the cable to be 1174 - 576 = 15.19 m (49.8 ft). The minimum cross-section would be that to satisfy the pallet only payload configuration (see Table 2-17) and would weigh 6.7 Kg/M (4.5 lb/ft). The shortest length of cable required would be 690 - 576 = 2.9 m (9.5 ft). Therefore, 12.29 m (40.3 ft) of cable would be required to be stowed which is possible but not very practical. In addition, the excess cable would weight 82.3 Kg (181.4 lbs) which again appears to be impractical.

The cable lengths used here, and in the balance of the study, are "x" station to station only and do not include any cable from the Orbiter payload panels to the wire tray nor the cable from the wire tray to the payload. These lengths are assumed to be approximately the same for all payloads. As such, they are further assumed not to affect the concept evaluation nor selection.

The current design practice on the Orbiter program is to permit no electrical connectors within a wire tray because of the space problem (connectors are large compared to cables) and also to minimize complications in the event troubleshooting is required along a cable. (No connector - no problem). Thus, option 5 was also discarded. The remaining options 1, 2, and 4 will be discussed in the next task.

Task 3 Options Evaluation

The use of a separate cable for each payload and each location (Option 1), as is the current planned practice, has the obvious advantages of minimum weight, signal loss and EMI problems because it has been optimized for a particular application. However, this optimization creates a very limited

versatility or capability to satisfy other payloads requirements and thus increased (compared to a concept designed for multiuse) kit installation and checkout time. This is because of the need to remove the cable set after every mission. The major disadvantage of this concept is the cost of developing and manufacturing a set of cables for every, or almost every, mission for each payload.

Option 2 will be a piece of MMSE and as such will reduce (compared to Option 1) the handling, installation and checkout effort and because of the fewer cables required, a significant cost saving. Since many applications of an MMSE cable set will not require the full length of cable, the signal losses and distortion will be greater than with the use of an optimum designed cable. In addition, a small weight penalty and potential excess cable stowage problem can be expected. The Option #2 concept can be designed to satisfy a pre-selected set of payload bay locations (2A) or to have the capability of supporting any location within the bay (2B). However, greater capability (mission flexibility) can only be achieved at the expense of additional cables or greater excess cable length, with the attendant electrical losses, for most payloads. The significance of excess weight for payloads versus lower total program cost requires effort beyond the scope of this study but is required before the final optimum MMSE configuration can be selected.

The use of a permanently installed cable with multiple outlets along the bay (Option 4) provides high commonality, the minimum handling and installation effort and with the proper selection of the short payload-to-wire-tray cables, unlimited payload location support capability. The problems of this concept, when compared to any of the others previously discussed are maximum excess weight and electrical losses for all payloads. An additional problem, but not expected to be significant, is that of EMI because of the long length of unused cable and unused connectors. As with Option 2, a trade between excess mission weight and program cost are required before the real value of this option can be assessed.

In order to provide a quantitative means of evaluating the four remaining options, two factors will be calculated for each option. The first is a cost factor which is based on the estimated weight of a complete set of cables. The weight is a function of the length and cross-section. The weight factor (non-dimensional) of the various types of wire is as follows:

Twisted Shielded Pair - TSP = 0.0156/unit length
Twisted Pair - TP = 0.00725/unit length
Coax - Cs = 0.01/unit length
Twisted Shielded Five Conductor - TS5C = 0.0244/unit length

The second factor is a mission weight factor. This is based on the estimated weight of the cable required to support a worst case payload configuration. For this analysis, the total Orbiter payload was assumed to be made up of an IUS and NASA payload with the IUS interface at Station 1128 and a multiple pallet spacelab payload with its interface at Station 690.

The cost factor for Option 1 was calculated to be 1476.2. The details for this analysis are listed in Table 2-19. The data for this and the succeeding analyses are taken from Table 2-17 and Figure 2-34. The analysis for the cost factor for Option 2A is shown in Table 2-20 and for 2B in Table 2-21. The factor for Option 2A is 798.2 and for 2B is 866.5. The cost factor of 331 for Option #4 is based on a cable with the total capability of the wire requirements for a pallet configuration plus a NASA-IUS payload (87 TP, 264 TSP, 68 TS5C and 23 Cx).

The mission weight factors analysis is summarized in Table 2-22.

The quantative comparison of the four options was accomplished by first normalizing all cost and mission weight factors to Option 1, which is the current program plan. The data shows Option 4 to be one fourth the cost of the baseline, Option 1, but the mission weight factor is 2.3 (mission cable weight 2.3 times weight of Option 1 cable configuration). Excess weight could be unacceptable to some payloads. The costs of Option 2 are expected to be about half of Option 1 (cost factors 0.54 and 0.59) but, again, the excess weight for these options may not be acceptable.

The normalized cost and mission weight factors are listed in Table 2-23.

The schedule of Figure 2-35 reflects the assumption that no support will be required during OFT. The first need for the lines is the first operational flight (#7) with an assumed six month lead time between delivery and first usage. The schedule indicates that the pre-phase A effort (Requirements and Concept Reverification) need not start until 1978, thus providing adequate time for the necessary payload detailed requirements to be accumulated.

Conclusions

The study has identified three cable set concept configurations which show good promise for use as MMSE with cost savings which are expected to be significant. The final configuration of the cable set(s) will be dependent on the results of an excess weight/program cost trade and other considerations. Some of these considerations are cable routing, stowage and EMI. (One item which would have a significant effect on cable configuration is the use of an MDM-Data Bus concept for the payload to Orbiter data transmission.)

Recommendations

During the study, several items of concern or uncertainty have been identified which are recommended for inclusion in any future study:

1. Impact of excess weight and length on payload ground and flight operations.
2. Cable routing and support.
3. Effects of mixed payloads on the commonality of cables.
4. Potential EMI problems with stowage of excess cable, unused connectors (Option 4) and long unused cables (antenna).

Table 2-19. Option 1 Cable Set Cost Factor Analysis

Payload	Sta	Cable Length	TP No.	TP CF	TSP No.	TSP CF	TS5C No.	TS5C CF	CX No.	CX CF	TOTAL No.	TOTAL CF
Pallet	694	118	62	4.5	173	27.0	48	11.7	18	1.8	728	45.0
	814	238	62	9.0	173	54.0	48	23.3	18	3.6	728	89.9
	934	358	62	13.5	173	81.0	48	35.0	18	5.4	728	134.9
	1054	478	62	18.0	173	108.0	48	46.7	18	7.2	728	179.8
	1174	598	62	22.5	173	134.9	48	58.3	18	9.0	728	224.7
Module and Pallet	756	180	17	1.8	102	23.9	40	14.4	12	1.8	450	41.9
Module	815	239	17	2.5	102	31.8	40	19.4	12	2.4	450	56.1
DOD/IUS	833	257	25	3.9	75	25.0	20	10.4	3	.6	303	39.9
	951	375	25	5.6	75	36.6	20	15.2	3	.9	303	58.4
	1010	434	25	6.5	75	42.1	20	17.5	3	1.1	303	67.2
	1128	552	25	8.3	75	53.8	20	22.3	3	1.4	303	85.8
NASA/IUS	833	257	25	3.9	91	30.4	20	10.4	5	1.1	337	45.7
	951	375	25	5.6	91	44.4	20	15.2	5	1.6	337	66.9
	1010	434	25	6.5	91	51.1	20	17.5	5	1.8	337	75.9
	1128	552	25	8.3	91	65.3	20	22.3	5	2.3	337	98.2
DOD Free Flyer	710	134	←				TBD					→
	1069	493										
LCMS	1069	493	87	25.9	183	117.0	20	20.9	5	2.0	650	165.9
CF = Cost Factor										TOTAL		1476.2

Table 2-20. Option 2A Cable Set Cost Factor Analysis
(6 Cable Configuration - Identified Payload Stations Only)

Cable Number	Frame Exit (STA)	Payload Interface Stations Serviced	Length	Cable Cross Section and CF								Total CF
				TP		TSP		TS5C		CX		
				Number	CF	Number	CF	Number	CF	Number	CF	
1	693	694 710	134	62	5.0	173	30.1	48	13.1	18	2.0	50.2
2	750	756	180	17	1.9	102	23.9	40	14.6	12	1.8	42.2
3	807	814 815 833	257	62	9.6	173	57.8	48	25.1	18	3.9	96.4
4	980	934 951 1010	434	62	16.3	173	97.6	48	42.3	18	6.5	162.7
5	1090	1054 1069 1128	552	87	29.0	183	131.4	48	53.8	18	8.3	222.5
6	1141	1174	598	62	22.4	173	134.5	48	58.3	18	9.0	224.2
											TOTAL	798.2

CF = Cost Factor
Orbiter Interface = STA 576

Table 2-21. Option 2B Cable Cost Factor Analysis
(Service All Stations)

CABLE NO.	FRAME EXIT (STA)	STATIONS SERVICED	TOTAL LENGTH	LENGTH OUTSIDE TRAY	CABLE CROSS SECTION & CF								COST FACTOR
					TP		TSP		TS5C		CX		
					NO.	CF	NO.	CF	NO.	CF	NO.	CF	
1	750	694 – 806	230	56	62	8.6	173	51.8	48	22.4	18	3.5	86.3
2	863	806 – 920	344	57	62	12.9	173	77.4	48	33.5	18	5.2	129.0
3	980	920 – 1040	464	60	62	17.4	173	104.4	48	45.2	18	7.0	174.0
4	1090	1040 – 1140	564	50	87	29.7	183	134.2	48	55.0	18	8.5	227.4
5	1191	1140 – 1242	666	51	62	25.0	173	149.9	48	64.9	18	10.0	249.8
												TOTAL	866.5

Table 2-22. Mission Weight Factor (MWF)

Option	Payload Type	Payload Station Location	Mission Weight Factor	Total MWF
1	NASA-IUS Pallet	1128 694	98 45	143
2A	NASA-IUS Pallet	1128 694	222.5 50.2	272.7
2B	NASA-IUS Pallet	1128 694	227.4 86.3	313.7

Table 2-23. Cost and Weight Factors Summary

Option	Cost Factor	Normalized Cost Factor	Mission Weight Factor	Normalized MWF
1	1476	1	143	1
2A	798	.54	273	1.9
2B	867	.59	314	2.2
4	331	.22	331	2.3

SCHEDULE CONSIDERATIONS

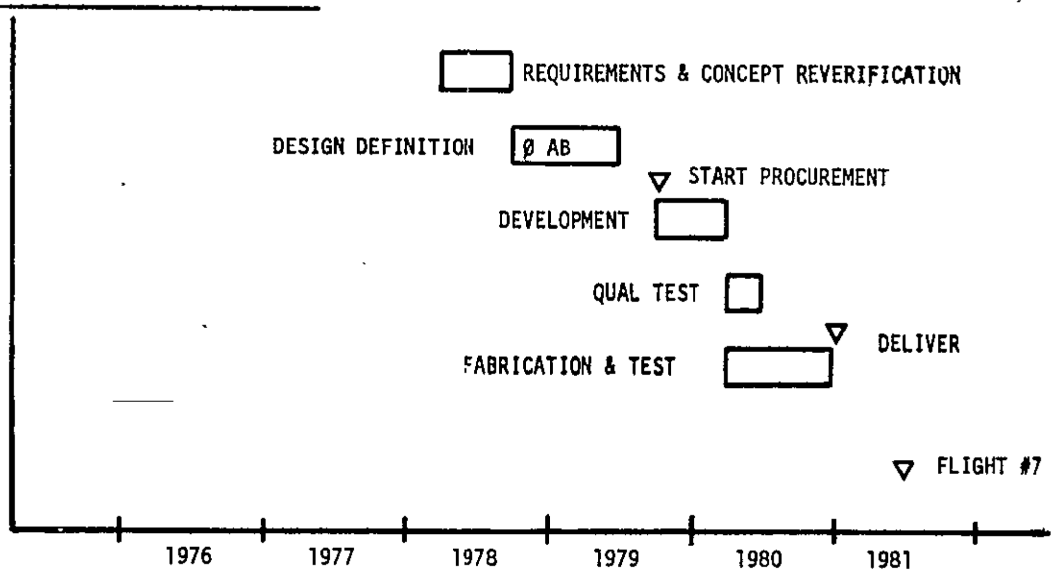


Figure 2-35. Implementation Schedule

5. Total inventory of cables (mission and ground operations dependent).

6. The effect on the number and configuration of MMSE cable sets if a combination of common and payload unique cables is used.

7. Use of the MDM Data Bus concept (described elsewhere).

PAYLOAD/ORBITER FLUID LINES

Introduction

The total program cost associated with each payload being responsible for the design and procurement of their fluid servicing lines (Orbiter payload bay interface to payload interface) would be necessarily high. Ideally, the design, fabrication, and utilization of a set of lines to satisfy all payloads, regardless of their location in the bay, should result in considerable cost savings. Even if only a partial or limited degree of commonality is achievable, the savings are likely to be worthwhile. The lines in question are utilized for coolant flows, pressurants, purge gases, propellants, etc.

Objective of Task

The objective of this study was to make a preliminary determination of the degree of potential commonality and to define the most promising configuration of the Orbiter-to-payload servicing fluid lines.

Approach

To accomplish the study objective, the effort was broken into three tasks. (1) Define the fluid requirements for all payloads having fluid interfaces with the Orbiter, considering the interface connections, fluid media, line size and pressure levels; (2) determine and evaluate the options available to satisfy the requirements and identify the fluid compatibility groups; and (3) indicate the most promising concept configuration(s) by defining the different lengths for each chosen fluid grouping.

Results

Task 1 - Requirements Identification

Investigation of the SSPD, Spacelab, DoD and TUG documentation resulted in the identification of 14 gases and liquids which must be supplied to the various payloads through the Orbiter umbilicals and/or payload bay interface panels. These fluids were segregated into compatibility groups as noted below:

Group 1 - Freon, H_2O
Group 2 - AR/CH_4, CH_4, GHe, GN_2, GH_2, Xe
Group 3 - LHe, LH_2, LN_2
Group 4 - LO_2
Group 5 - GO_2
Group 6 - N_2H_4

Freon and H_2O were put into the same group because they are both used for the heat exchange fluid between the Spacelabs and the Orbiter thermal control system. Freon is used for the pallet only configuration when no personnel are involved. Water is used whenever a module is a part of the payload configuration. Both fluids use the same passages in the Orbiter payload heat exchanger. Gases and liquified gases (cryogenics) were put into separate

groups not because they were incompatible, but because of the potential design problems on using a common line. The gas lines will no doubt be high pressure $\approx$ 20700 KPA (3000 psig) and the cryo lines must be vacuum jacketed. Trying to combine both these requirements into a single line was expected to unnecessarily complicate the design and handling procedures. Oxygen and hydrazine were kept separate for the obvious reasons of safety. All lines will be purged and/or cleaned after each mission depending on their next usage.

Most of the fluid requirements are for filling of payload storage tanks and therefore some fluids will require more than a single line. For this study, it was assumed that only the cryo lines require a vent line. The Freon and water lines are for payload cooling so they also require two lines, one supply and one return. It was assumed that these lines would always interface with the Spacelab at its forward face regardless of configuration. The other fluid lines may have to interface with Spacelab pallets anyplace in the bay as the basic Spacelab design does not include the capability to transfer fluids between pallets if they are not structurally attached. Discussions with Shuttle design personnel indicate that it is good design practice to make all "B" nut type fluid connections within 12 inches of a point of attachment to the structure. This practice minimizes the potential for leakage as a result of Shuttle vibration.

In addition to the fluids themselves, other requirements pertaining to fluids were identified. They include the type of payload (i.e., Spacelab pallet, module plus pallet, module only and automated), number of missions (usage frequency), line size and the desired servicing time. The identified countdown servicing time dictates the ground-to-Orbiter interface panel and thus the Orbiter-to-payload interface within the bay. The total requirements for the fluids have not been identified by the respective payload disciplines at this time, thus the absence of any pressure data and only very little line size data. The total requirements available for those payloads identifying the need for fluids are shown in Tables 2-24, 2-25, and 2-26.

The respective payload documentation identified the required fluid interface stations of the installed payloads. In the case of the pallet only configuration of the Spacelab, ERNO identified Station 710 as the fluid interface for a 5 pallet configuration and Station 714 for a 3 pallet configuration. Similar single interface stations were identified for the module plus pallet and module only configurations. Actually, the payloads can be placed almost any place in the bay depending on the other payloads on a particular mission and the center of gravity considerations. Since only three pallets can be structurally attached and up to five pallets can be flown on a single mission, the fluid interface for a pallet can be at any one of the five pallet locations in a five pallet mission. Thus, five potential interface locations were used for this study. The identify of all payload interface locations used is shown in Figure 2-36. This figure also shows the location of the Orbiter-to-payload interface panels and the fluids the payload requires from each. A summary of the payload fluid requirements by payload type and Orbiter interface station is found in Table 2-27.

Table 2-24. Payload Type Fluid Requirements

Payload Type	Fluid	Line Size cm (in)	Payload Interface Station	Orbiter Interface Station
IUS	No fluid	requirements.		
LDEF	No fluid	requirements.		
DoD	GN_2	0.6 (0.25)	720	1307
Spacelab				
Module	H_2O	1.9 (0.75)	815	586
	GO_2	0.6 (0.25)	815	586
Module + Pallet	H_2O	1.9 (0.75)	748/756	586
	GO_2	0.6 (0.25)	748/756	586
Pallet	Freon	1.9 (0.75	690/694	586
Tug	LO_2 F&D	5.1 (2.0)	1246	1307
	LO_2 Dump	12.7 (5.0)	1246	1307
	LO_2 Top	1.9 (0.75)	1246	1307
	GH_e Vent	1.9 (0.75)	1246	1307
	GH_e Fill	1.0 (0.375)	1246	1307
	LH_2 F&D	5.1 (2.0)	1246	1307
	LH_2 Dump	12.7 (5.0)	1246	1307
	LH_2 Top	5.1 (2.0)	1246	1307
	LH_2 Vent	7.6 (3.0)	1246	1307
	LH_2 Relief	6.4 (2.5)	1246	1307
	N_2H_4 F&D	1.0 (.385)	1246	1307

Table 2-25. Individual Spacelab Payloads - Fluid Requirements

Payload	Type	Fluid	Payload Interface Station	Orbiter Interface Station
AS-01-S	3B	LHe	690	835
AS-03-S	2P	GN_2	690	835
AS-14-S	1P	LHe		
AS-15-S	4P	LHe		
AS-18-S	4P	LHe		
AS-20-S	2P	LH_2, LHe		
AS-31-S	4P	LHe		
AS-32-S	4P	LHe		
AS-51-S	2P	LHe		
AS-54-S	4P	LHe		
AS-61-S	1P	LHe		
HE-11-S	2P	GN_2, AR/CH_4		835
HE-15-S	2P	LHe		1307
HE-21-S	1P	LHe		835
HE-22-S	1P	LHe, Xe, CH_4	690	
LS-09-S	M	GN_2, LH_2, GH_2	815	
SO-01-S	5P	GN_2	690	
SO-11-S	3P	GN_2		
SO-13-S	4P	LHe		
SO-14-S	2P	LN_2		
SO-15-S	4P	GN_2		
SO-17-S	2P	GN_2	690	
SP-31-S	M	LN_2	815	
EO-11-S	M+P	LN_2	748	
EO-12-S	M+P	LN_2	748	
ST-59-S	M	LHe, GHe	815	
CN-05-5	M+P	LN_2	748	835

Table 2-26. Automated Payloads - Fluid Requirements

Payload	Fluid	Payload Interface Station	Orbiter Interface Stations
HE-08-A	TBD	TBD	835
HE-09-A	LHe	TBD	835
HE-11-A	TBD	TBD	TBD
HE-12-A	TBD	TBD	TBD
SO-02-A	TBD	TBD	835
SO-06-A	LN_2	TBD	835
AP-04-A	LHe	TBD	835
AP-06-A	LHe	TBD	TBD

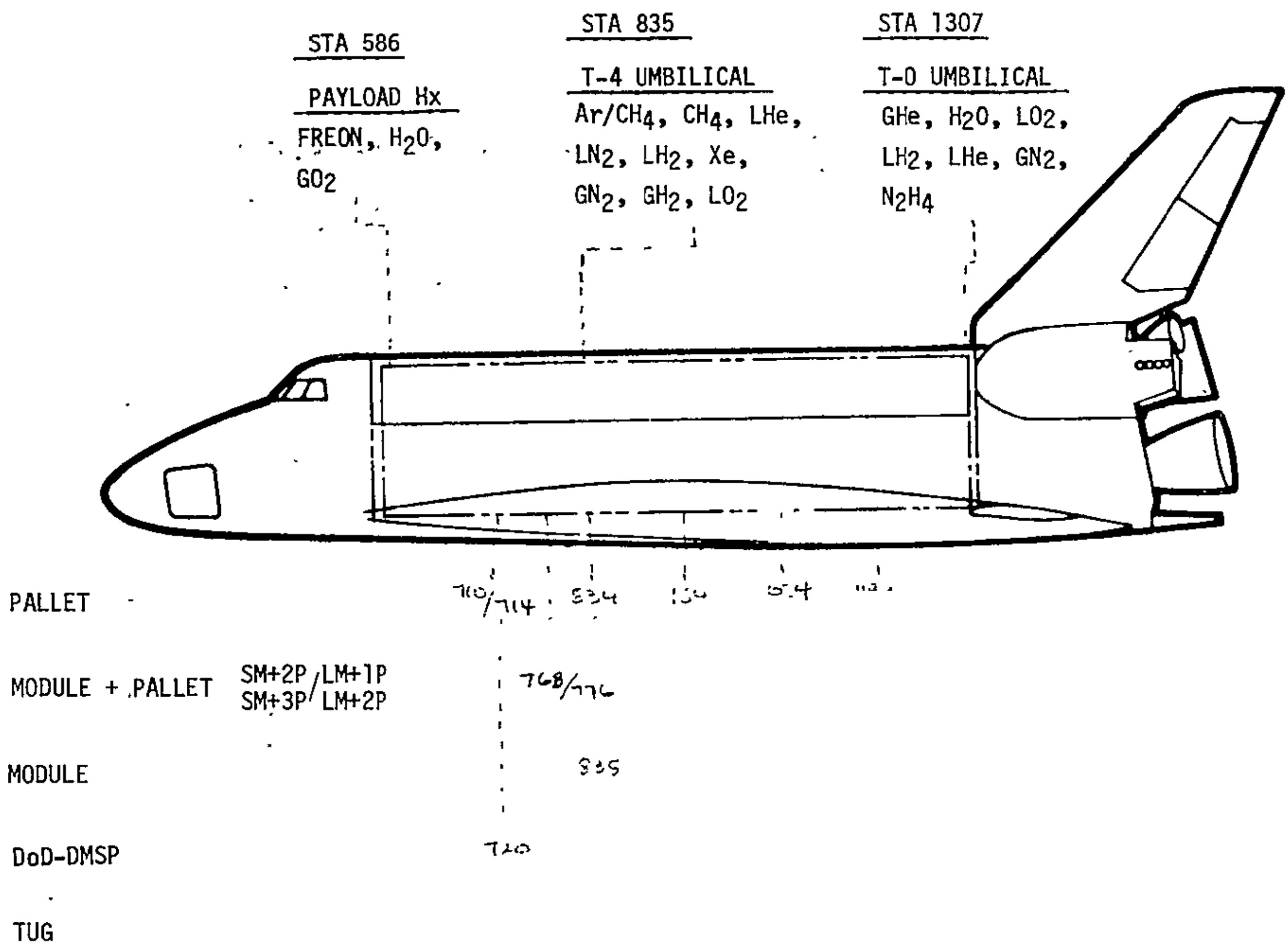


Figure 2-36.—Orbiter and Payload Fluid Interface Locations

Table 2-27. Summary of Payload Fluid Requirements

	ORBITER INTERFACE																		
	STA 586			STA 835									STA 1307						
PAYLOAD TYPE	GO_2	Freon	H_2O	Ar/CH_4	CH_4	GN_2	GH_2	Xe	LN_2	LH_2	LHe	LO_2	GHe	GN_2	LHe	LH_2	H_2O	LO_2	N_2H_4
SPACELAB																			
PALLET		XX		X	X	X		X	XX	XX	XX	XX			XX				
MODULE + PALLET	X		XX	X	X	X		X	XX	XX	XX	XX							
MODULE	X		XX			X	X		XX		XX								
DoD-DMSP														X					
TUG													X			XX	X	XX	X
FLUID COMPATIBILITY GROUP	5	1	1	2	2	2	2	2	3	3	3	4	2	2	3	3	1	4	6

Task 2 - Concept Determination and Evaluation

Three options were identified for consideration and are discussed in the following paragraphs.

Option #1 uses a separate line for each fluid and location and is the implicit approach currently taken on the program. This option provides perfect compatibility for the payload for which it was designed, but provides no commonality with any other payload. This option imposes the maximum cost for manufacture and provides the capability for servicing of the payload only at a single location within the bay. Each location would require a new line.

Option #2 uses common lines for compatible fluids for each location. This option provides some commonality for compatible fluids with common interface points. Manufacturing costs would be reduced because common lines would allow multi-use. In instances where subsequent mission payloads utilized the same interface station with a compatible fluid, the removal/reinstallation/checkout cycle would be eliminated. This option would impose a small weight penalty on payloads for which smaller line sizes might be adequate (as the lines would be sized to support the larger payload requirement. This option basically only provides the capability for support of the payload at a single location, however some flexibility exists because of availability of more than one length for a fluid because of commonality between fluids.

Option #3 uses common lines for compatible fluids and for multiple locations with the hard line made up of modular sections. This option provides the maximum amount of commonality with a minimum amount of lines. Manufacturing cost would be at a minimum because the required payload lines will be "made up" from several different lengths of line (each length is a multiple of the shortest). A common length flex line, for connecting to the payload, will be used with each hard line. Using this concept, unlimited payload location options are permitted as well as simplified handling and storage of the lines. Connection of the necessary lengths to make up the complete line can be accomplished by two methods. Dynatube mechanical connections can be utilized for ease of installation but this method presents potential leak problems. The second and probably the most desirable is to use brazed joints. These can be "disconnected" (unbrazed) and the component line lengths reused in another line assembly. This method increases the line assembly time, as compared to a Dynatube joint, but practically eliminates the leakage problem. The actual method of connecting the short lengths of line, of handling them during assembly on the bench and in the Orbiter and the method of storing, will be resolved in a subsequent, more detailed study effort. A simple sketch of each option, along with a summary of its advantages and disadvantages, is shown in Figure 2-37.

Task 3 - Selection of Most Promising Option

The evaluation of the options resulted in the selection of Option #3 as the most promising because it has the greatest potential for commonality and can provide complete coverage of the payload bay. It thus eliminates any restrictions on payload location from the standpoint of cost of providing the necessary fluid servicing lines.

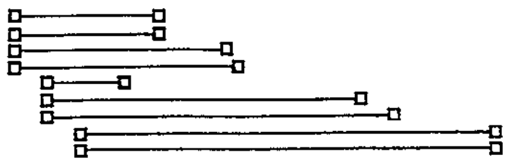
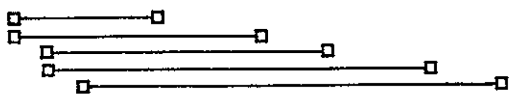
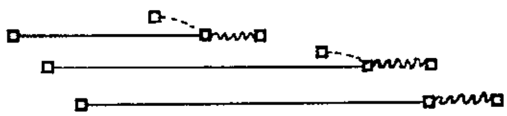
OPTION	CONFIGURATION	ADVANTAGES	DISADVANTAGES
1. SEPARATE LINE FOR EACH FLUID AND LOCATION		• NO WEIGHT PENALTY	• NO COMMONALITY; LARGE NO. OF LINES • LIMITED PAYLOAD LOCATION OPTIONS
2. COMMON LINE FOR COMPATIBLE FLUIDS FOR EACH LOCATION		• SOME COMMONALITY	• SMALL WEIGHT PENALTY FOR COMMON LINE SIZE • LIMITED PAYLOAD LOCATION OPTIONS
3. COMMON LINES FOR COMPATIBLE FLUIDS AND FOR MULTIPLE LOCATIONS (MODULAR HARD LINES)		• SIGNIFICANT COMMONALITY • UNLIMITED PL LOCATION OPTIONS	• MODERATE WEIGHT PENALTY (COMMON LINE SIZE, FLEX SEC. & EXTRA LINE LENGTH) • MULTIPLE JOINTS

Figure 2-37. Fluid Line Configurations Options

Selection of the length for the flex section is relatively straight forward. An investigation of the midbody frame location shows that if the flex line is one-half the maximum spacing between frames than it will be capable of covering any location in the bay with the correct number of pieces of hard line attached. The maximum frame spacing is 153.8 cm (60.5 in), therefore, a 77.5 cm (30.5 in) length of flex line was selected. The lengths of hard line must be determined by an economic and practical evaluation of the number of different lengths versus the cost and time to make up a complete line (number of joints). The line lengths are also a function of the ease of handling and storing. A modular length of approximately 152 cm (60 in) appears to be a reasonable starting point for a more detailed investigation because of its compatibility with the Orbiter mid-body frame spacing and the ease with which it could be handled and stored. The Orbiter mid-body frame locations and spacing can be found in Table 2-28. The above discussion applies to X-length, single-plane dimensions only. The three 3-dimensional lengths of flex line in particular must be determined later with extensive layouts. As shown in Table 2-26, very little data is available on the automated type of payloads. However, the versatility of the concept discussed above is expected to be sufficient for satisfying the automated payload fluid requirements.

Each of the fluid compatible groups will require a number of hard and flex lines in the inventory to permit "build-up" of the correct line length and type for each mission.

Table 2-28. Orbiter Midbody Frame Location & Spacing

Station	Distance Between Station Centimeters	(Inches)
592		
	137.2	54
636		
	144.8	57
693		
	144.8	57
750		
	144.8	57
807		
	142.2	56
863		
	142.2	56
919		
	153.7	60.5
979.5		
	153.7	60.5
1040		
	127.7	50.3
1090.3		
	127.7	50.3
1140.7		
	127.7	50.3
1191		
	147.3	58
1249		
	147.3	58
1307		

It may be cost-effective to build permanent longer sections of hard line if later investigation of the traffic model indicates a large number of flights requiring a particular length. (However, the installation and handling complications with long lines must be considered.)

While the length of line for any option can be determined by analyzing the interface station-to-station requirements, the lack of data on line diameter and operating pressures prevents a complete definition of the required line set. It will be necessary to perform a cost and weight trade to determine the optimum number of lines and size of lines. It may not be practical, for instance, to design the same line for the maximum pressure and diameter within a fluid group.

Potential changes being considered for the Orbiter can also affect the design of the lines and in the case of one stating "no liquid helium through an umbilical", even the concept to some degree.

Implementation Schedule

The schedule shown in Figure 2-38 reflects the assumption that no support will be required during OFT. The first need for the lines is the first operational flight (No. 7). With an assumed six month lead time between delivery and first usage, the figure indicates that the pre-phase A effort (Requirement and Concept Reverification) need not start until 1978, thus providing adequate time for the necessary payload detailed requirements to be accumulated.

Conclusions

A large number (approximately 119) of different lines would be required to satisfy the fluid requirements of all payload types at all the identified payload locations in the bay (5 payload types, 9 payload locations, 19 fluid interface locations). This number can be reduced by approximately 50% by doing nothing more than using a common line for compatible fluids. Further savings can be realized by use of a small number of different length lines from which the required line lengths can be assembled. Refinement of the concept requires additional detailed study and more complete payload requirements.

Recommendations

In the course of the study the following items have been surfaced which are recommended for consideration as items of concern or uncertainty in any future study (such as the recently directed design of these lines and cables for Spacelab):

1. Requirements for fill, vent and pressure relief lines.

2. Fluid line sizes, support configurations and routings.

3. Effects of mixed payloads on line commonality.

4. Line weights and cost comparisons for different operating pressures and diameters.

5. Total inventory of lines (based on traffic model and ground operations scheduling).

6. High probability of future requirements which must be satisfied, such as:

   a. Nitrogen from Orbiter (STA 586) to Spacelab
   b. RTG ground cooling (H_2O from Sta. 1307).

SCHEDULE CONSIDERATIONS

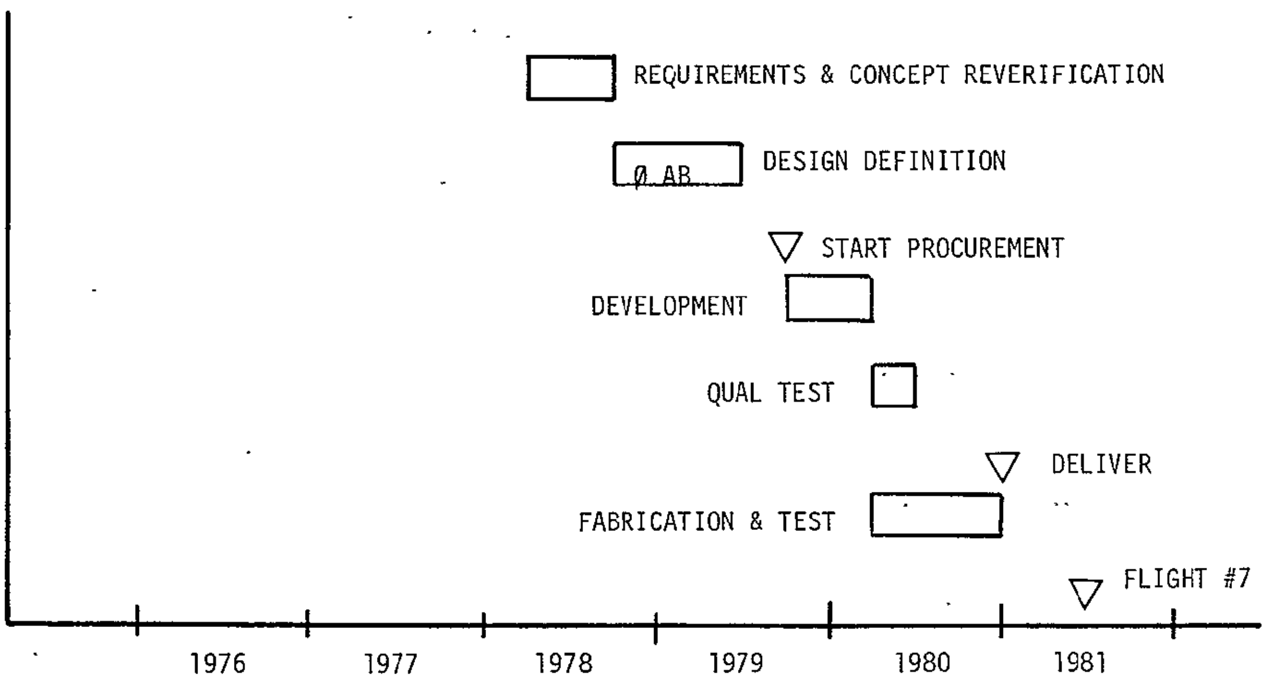


Figure 2-38. Implementation Schedule

7. Potential Orbiter changes, such as

   a. No LHe through umbilicals (drag-in lines through open payload doors or load before installation).

   b. New payload umbilical at Sta. 1278, primarily for liquids; T-4 umbilical primarily for gasses.

   c. Sta. 586 panel to Sta. 636.

MULTI-DISCIPLINE AUXILIARY PAYLOAD POWER SYSTEM (MAPPS)

Introduction

The Space Processing Application (SPA) discipline has for several years been showing the need for power levels greater than that available to the payloads from the Orbiter. Since this was a payload-unique requirement the auxiliary power was to be provided by the SPA payloads themselves. In the 1975 version of the SSPD several other payload disciplines have also indicated the need for power above the 4.0-5.2 kw net (plus 1.8 through 3.0 kw required for Spacelab systems) available to the Spacelab payloads or experiments. In addition, there are logical combinations of payloads which will require an auxiliary power system. The conceptual design of the MAPPS concept arose from a company-sponsored effort originally intended to provide a cheaper, more integrated alternative to the MSFC Auxiliary Payload Power System (APPS) concept. The work has been utilized here to define the special emphasis MMSE item.

The technical discussion of the company-sponsored study will be found in Appendix A4 of this report. Figure 2-39 presents a simplified cost statement and the implementation plan for MAPPS.

COSTS (FIRST UNIT)	$ 3,700,000
Concept Refinement Study	100,000
Detailed Design Study (Phase AB)	300,000
Delta Development	350,000
Fabrication	2,600,000
Test	350,000

SCHEDULE CONSIDERATIONS

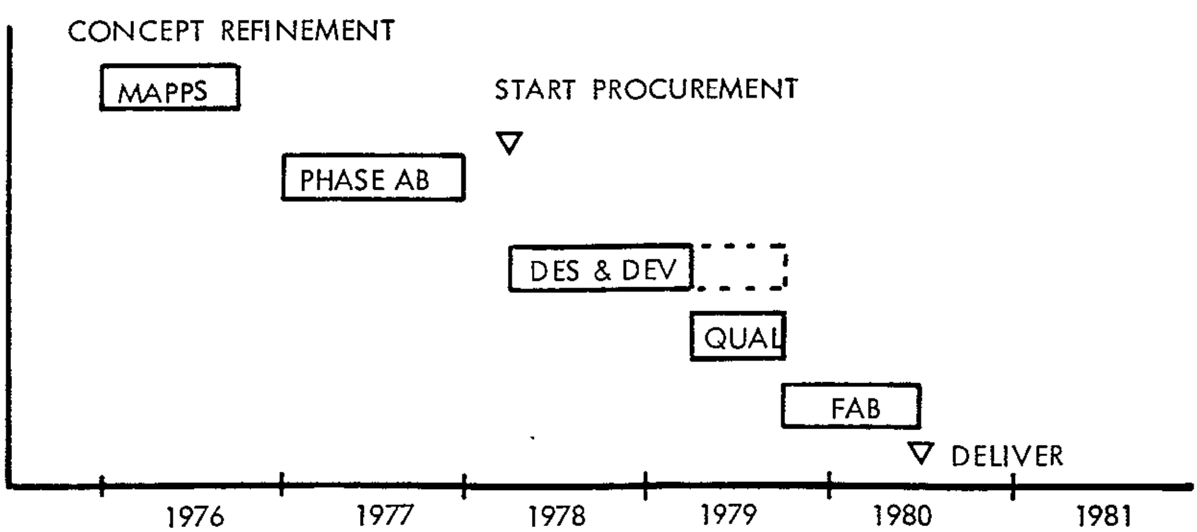


Figure 2-39. Cost and Implementation Plan

The development cost of $750,000 is low because none of the major components require development effort. The fuel cells, cryo tanks, flash evaporator, power conditioning equipment and radiators are all being developed for the Orbiter. The only items with a significant development cost are the keel fittings (modified) used for mounting of the fuel cells and auxiliary equipment.

The MAPPS kit is required to support the first flight of the AMPS payload early in 1981. Delivery is desired approximately six months prior to the first flight so that adequate time is available for in-house testing of the flight unit prior to its first usage. With this delivery date it is necessary to start a pre-phase A study (concept refinement) early in FY 1976.

Conclusions

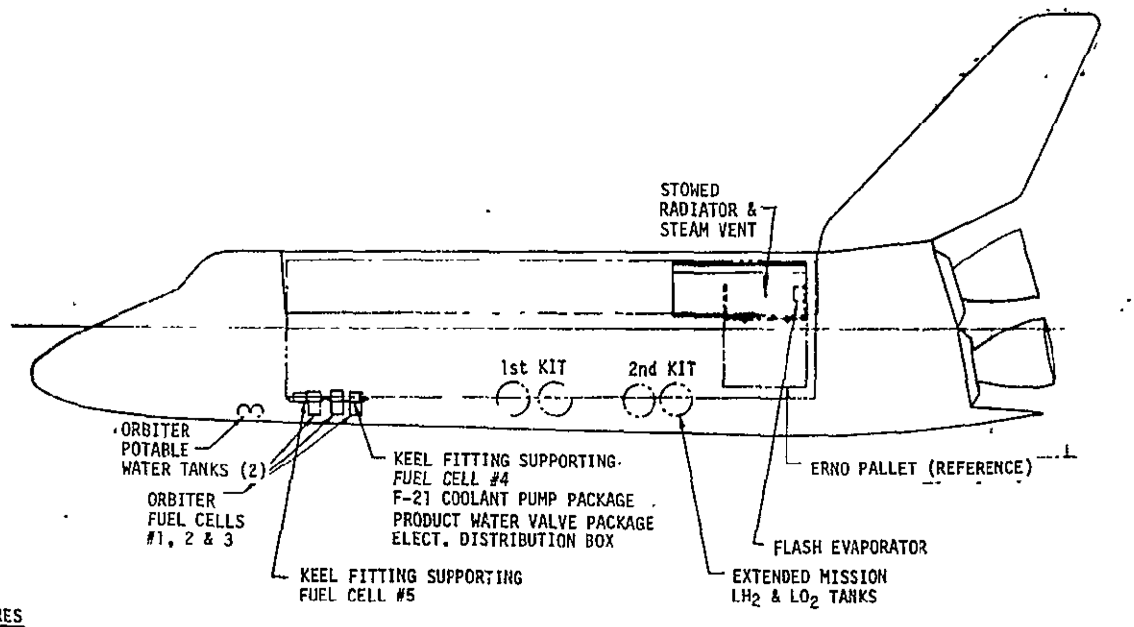
The MAPPS (or equivalent) is properly designated as "MMSE". There is a definite use for a multi-discipline auxiliary payload power system as evidenced by the 80 missions, identified in the SSPD, which require more power than can be supplied by the Orbiter.* The, as yet, undefined combined payloads further justify the need for an MMSE kit to satisfy their power requirements.

The MAPPS configuration shown in Figure 2-40 provides the needed capability at a very low development cost of $750,000 and low unit cost of $2,950,000. Its system weight is over 1500 kg (330 pounds) lighter than the similar APPS and only requires at most 76 cm (30 in.) of payload bay volume.

Recommendations

As a result of the conclusions noted above, it is recommended that the Rockwell concept be considered by NASA as an MMSE item and that a study effort be funded to further define the MAPPS concept at least to the level of depth of the MSFC APPS concept, which it would supplant for space processing missions.

*NOTE: ESA-Rockwell interface discussions in mid-December 1975 indicated that the requirements for Spacelab subsystems are even higher than described on the previous page.



FEATURES

1 - DESIGNED AS MMSE POWER GENERATION KIT

2 - UTILIZES MAX ORBITER EQUIPMENT
  - FUEL CELLS, RADIATORS, CRYO KITS, WATER TANKS, FLASH EVAPORATOR

3 - NO SPECIAL PALLET REQUIRED

4 - SMALL LOSS OF PAYLOAD BAY VOLUME (~4%)

5 - FLEXIBLE FOR VARIOUS POWER/ENERGY LEVELS
  - 1-5 CRYO TANK KIT
  - 1 or 2 FUEL CELLS

6 - SHORT CRYO LINES

7 - LOW WEIGHT (4327 LB)

8 - LOW PROCUREMENT COST ($3M)

Figure 2-40. MAPPS Description

PROGRAMMATICS

Costs and program implementation schedules were individually developed for the special emphasis items in the preceding sections, partially satisfying Tasks 5 and 6 requirements of the Study Plan. This section carries the cost evaluation a step further to estimate potential savings by adopting the new MMSE approaches and completes the Tasks 5 and 6 requirements. The program implementation schedules and costs are combined in order to provide an overall program management perspective.

Combined Schedules

Figure 2-41 summarizes the key schedules from the previous section for the six special emphasis items. Four items, the MAPPS, PMDM, MOST, and Spin-up Mechanisms, require prompt initiation of in-depth concept and preliminary design studies if the initial need (IOC) dates are to be met with normal manpower levels and minimum risk-taking. (More details are in the item's respective sections, above.)

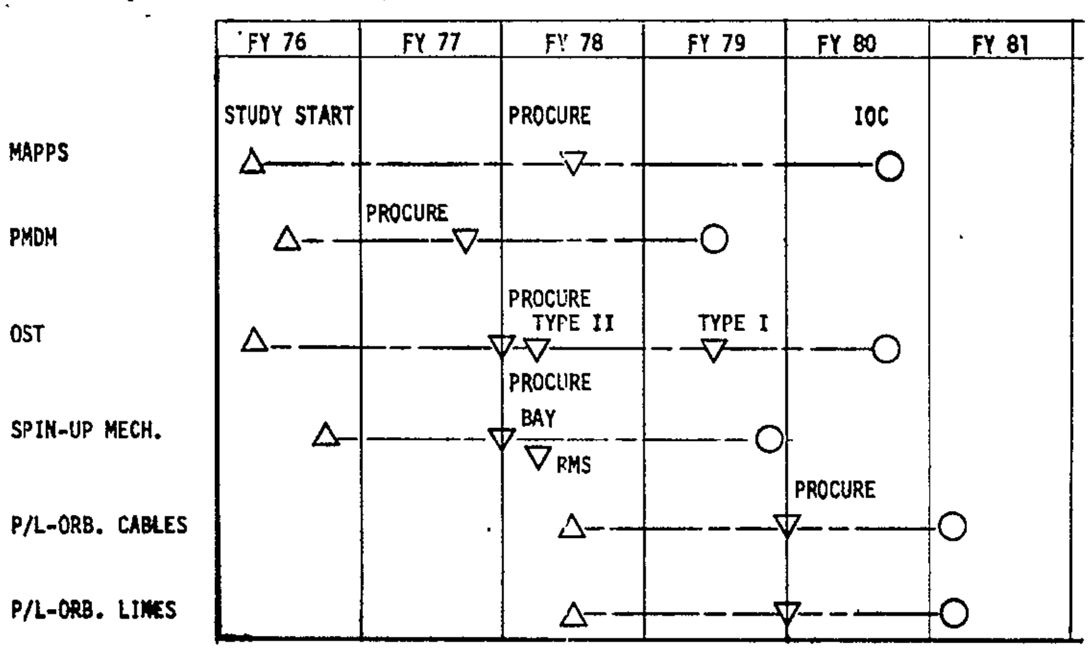


Figure 2-41. Summary of Timing for Recommended STS/MMSE Procurements

Three configurations of the MOST appear necessary. The greatest modification and test efforts are needed for the Type II and III configurations, where IOC dates are about mid-1980. Therefore, initial study and procurement action is primarily directed at MOST II and III. However, the concept study phase should also evaluate the status and promise of the new low cost star tracker, Star Tracker for Economical Long Life Attitude Reference (STELLAR), which uses a charge coupled device sensor and is under development by the Jet Propulsion Laboratory. This device is now in the "breadboard" development stage and could impact the procurement plans for MOST for one or more of the configurations.

The spin-up mechanisms both require early study and procurement to meet early need dates. The primary need for the RMS spin-up mechanism is to establish the RMS dynamic characteristics when used in this mode. The primary need for the bay platform spin-up mechanism is to further define and trade off design alternatives and investigate vibration environment compatibility with mounted payloads.

Procurements should be let for final design and hardware for the MAPPS, PMDM, MOST, and spin-up mechanisms by the end of FY 1977.

The MMSE fluid and electrical lines do not require immediate action. The primary investigative effort remaining is to optimize the basic design concepts. Considerably more data on specific payload requirements and bay location combinations are needed for this. Use of the PMDM may impact the electrical cables approach by minimizing (but not deleting) the number of cables. Lead time for design and fabrication is expected to be relatively short. Therefore, it is recommended that procurement be delayed even though MMSE thus may not be available for initial flights. Initial experience in OFT for custom installations can be applied to ultimate MMSE concept design.

Combined Costs

MAPPS Cost Savings. Eighty payload flights were identified as requiring auxiliary power exceeding Orbiter capability. Additional mixed or grouped payload combinations are sure to be encountered and will likely require auxiliary power but such potential combinations have not yet been identified. Of the 80 flights, all require more power than battery kits could provide due to total energy needs. Therefore, MAPPS is applicable to all 80 flights as a minimum.

An Auxiliary Payload Power System (APPS) is currently being studied by MSFC for Space Processing payloads in particular. APPS could potentially be used to support other payloads, although it requires more payload bay volume and weighs more, which probably means it could fly fewer payload missions than can MAPPS. For ROM cost savings purposes, however, it is assumed that APPS could provide the same support as MAPPS. MAPPS cost savings are, therefore, based upon comparison to APPS.

MAPPS development cost is small since no major new hardware is used; keel fittings being the only significant new components (and even they may "fall out" of the OFT program). The APPS requires new deployable radiator and pallet/structure developments and requires significant cryogenics kit and control hardware, all of which is not needed by MAPPS. There are other differences which are largely offsetting (e.g., water tanks versus flash evaporator). APPS system integration, management and fee will be considerably more due to higher cost of parts and more complex total system. Operation and maintenance is assumed to be comparable for the two concepts.

It is assumed that four units are required over 10 years, two for primary operation and two equivalent sets for spares or attrition.

C-2

The results are summarized below:

APPS		
Development	$ 3.6M	
Four units x $3.9M ea	15.6M	
		$ 19.2M
MAPPS		
Development	$ 0.6M	
Four units x $2.6M ea	10.4M	
		$ 11.0M
MAPPS Savings over APPS		$ 8.2M

Electrical Cabling Savings. Since there is no data on an existing MMSE cabling approach for comparison, cost savings were determined with respect to custom designs for all flights. Since each payload and mixed payload configuration will likely cause different line lengths and signal mixes, each flight would likely be a new design. For convenience, 500 flights are assumed, a compromise between minimum and maximum mission traffic models. (It has been argued that about 25 percent of the total flights for IUS would not require re-design cables for each flight. However uncertainty on IUS payload requirements is an issue.)

The principal cost elements for long, large cables is for design, test, fabrication and installation. It is assumed that installation would be comparable for custom or MMSE cables, a conservative assumption.

It was further assumed that 10 cables per MMSE set will be required and that two sets would be procured. Current study indicates that probably less than 10 cables per set will be needed so this is felt to be conservative with respect to MMSE savings.

It has been estimated that it costs at least $30,000 to design and test a large cable and $5,000 for fabrication and materials costs. MMSE cable costs will not exceed custom cable costs but could cost less for some cables. However, both were conservatively assumed to be comparable. The MMSE cables will need some maintenance after each flight, such as inspection, test and minor repairs (50 hours at $20/hr).

These assumptions results in the following estimate:

MMSE		
Design and Test, \$30K x 10 =	\$ 0.3M	
Materials & Fab, \$ 5K x 10 x 2 =	0.1M	
Inspection & Test after Flights, \$1K x 500 =	0.5M	
		\$ 0.9M
CUSTOM		
Design & Test, \$30K x 500 =	15.0M	
Materials & Fab, \$5K x 500 =	2.5M	
		17.5M
MMSE Savings Over Custom		\$16.6M

Fluid Lines Savings. Since there is no data on an existing fluid lines MMSE approach, cost savings were determined with respect to custom designs for all flights. Since each payload and mixed payload configuration will likely cause different line lengths and fluid mixes, each flight would likely require a new design. For convenience, 500 flights are assumed.

It is estimated that for the types of lines required, including vacuum jacketed, high pressure, and flex types, it will cost at least \$50,000 per installation. This could very well be low unless the same agency produces all designs to take advantage of experience gained. (MMSE savings will increase to the degree this estimate is low.) It was assumed that the cost of developing the MMSE is three times that of a custom design. Based on using standard lengths to assemble specific MMSE installations, determination of the optimum sizes is the only major engineering problems.

Most significant is the inspection, tests and repair/replacement of sections after flight. This was estimated to be 5 to 10 percent of a new installation in labor and materials.

The results are summarized below:

MMSE		
Design, Test, Fab	\$ 0.10M	
Inventory of Components	0.05M	
		\$ 0.15M
Inspect, Test, Repair/Replace-\$4K x 500 flts		\$ 2.00M
		\$ 2.15M
CUSTOM		
Design, Test, Fab, \$50K x 500		\$25.00M
MMSE Savings Over Custom		\$22.85M

PMDM Cost Savings. The PMDM concept was compared to the existing custom hardwire cabling concept. The cabling costs developed previously (electrical cabling savings) are utilized as one cost element. However, other significant system level hardware costs occur. A major cost occurs because most signals in long, large cables will require line driver amplifiers, calibration circuits, filters, etc., to account for losses, noise susceptability and characteristic variations of different interconnecting cabling. Cable variations can be expected from development tests through final Orbiter integration. It is judged that each signal requires a total cost of $500 for delta interface design, specs, tests and components which would not be needed for short signal line lengths applicable to PMDM usage. It was previously assumed that an average of 250 signals per total payload cargo occurs. Where payloads digitize their own signals, to solve the long cable problems, the average cost per signal will be much greater. The cost assumption is therefore probably conservative relative to MMSE savings.

Another long, large cable cost is for Orbiter integration. It is judged that 100 manhours at $20 per hour is required for installation and a like cost for calibration and fixes. These costs are also probably conservative.

The installation costs for the PMDM data bus will be small in comparison. At most, two coax cables will be needed with several payload tie-in junctions. It is assumed that installation costs are proportional to the signal line pairs, or 2/250 of the hardwire installation.

However, the PMDM has additional non-recurring and recurring costs. It is assumed that 50 units would be procured to allow about five per flight and a large portion of the remainder to be loaned to payloads for home-base development test purposes. Recurring post flight maintenance and checkout should be low. The estimate for maintenance includes repairs/replacements. A new field test set requirement is also assumed. Also, a computer/processor is needed to manage the PMDM's in operation. The computer must be programmed for each flight with PMDM addressing, sampling and priority routines. This should become relatively simple after the first few flights. However, 500 manhours per flight was allocated for such programming. The computer can likely be an existing micro-computer. Autonetics, for example, has military qualified avionics, MOS, expandable micro-computers on the shelf at very low cost compared to past generation computers. However, an arbitrarily conservative cost for developing and procuring two computers was assumed.

The results of the above considerations are summarized below:

Electrical Cabling (Custom)

P/L-Crew Station Interface Hardware, 500 flts x 250 signals x $500/signal	$68.0M	
Cable Procurement	17.5M	
Installation, test, fix, calibration, 500 flts x 200 hrs x $20/h4	2.0M	
		$ 87.5M

PMDM

Development	$ .28M	
Inventory 50 x $100K	5.0 M	
Special Test Equipment Procurement	.2 M	
Computer		
Development	0.5 M	
Procurement (2)	0.2 M	
Recurring Installation, 500 x $160	0.08M	
Recurring Programming, 500 x $10K	5.0 M	
Recurring Post-Flight Checkout, Repair 500 x $2K	1.0 M	
		$ 12.26M
PMDM Savings Over Hardwire		$ 75.2 M

MOST Cost Savings. The modified Orbiter star tracker MMSE concept is compared to the previously proposed gimballed star tracker MMSE concept. Basic cost data was developed in the section for MOST.

Although three configurations are needed for 64 flights, more than one set (in general two OSTS are required for one 3-axis attitude reference) of trackers of either the same or different configurations will generally be needed. This is due to more than one pointing platform and/or pallet per flight. The alternative of using one set of trackers with autocollimators providing the reference to other platforms appears more complicated and costly than using independent star tracker sets (however, this needs investigation). Relatively more configuration II trackers are needed because of the multiple higher accuracy applications. The configuration I MOST can operate strapdown on some missions where more than one course pointing payload can use the single tracker set.

From the above it was judged that four configuration I, four configuration III, and six configuration II MOSTS would be needed to support 10 years of missions.

The gimballed tracker is inherently much less stable and reliable; a number of sets of components must be produced in order to select one set to achieve high accuracy. However, only one per pointing application instead of two are needed. It was assumed that two per flight plus two spares, or four total would cover requirements. The maintenance and spare parts costs were assumed comparable. This results in conservative MOST MMSE savings since MOSTS are expected to require little calibration or repair after flights.

The original MMSE item also requires a new development high accuracy unit to accommodate requirements that the gimballed "coarse" tracker cannot handle. The new development is judged to cost about twice the cost of modifying an OST (MOST III). It is felt that a third tracker comparable to MOST II would also have to be developed because gimballed trackers of the type proposed likely cannot achieve stable five sec performance. However, since the third tracker was not previously identified, it was not costed. Therefore, the MOST MMSE savings are felt to be very conservative. The results are summarized below:

MOST			
Config. I	- Development	$ 0	
	- Recurring, 4 x 120K	0.420M	
Config. II	- Development	0.315M	
	- Recurring, 6 x 195K	1.117M	
Config. III	- Development	0.420M	
	- Recurring, 4 x 270K	1.080M	
			$ 3.35M
Gimballed & New Develop. MMSE			
Gimballed	- Nonrecurring	$ 1.4 M	
	- Recurring, 4 x 390K	1.56 M	
New Devel.	- Nonrecurring	1.0 M	
	- Recurring, 4 x 500K	2.0 M	
			$ 5.96M
MOST MMSE Savings Over Other MMSE			$ 2.61M

Spin Mechanism Cost Savings. There is no precedent for comparing either the bay or RMS MMSE spin mechanisms. Therefore, the number of new developments for the "current" concept assumes that commonality exists within families/user agencies.

Fifty bay platform spin deployments are assumed to be a compromise between a low of 44 and a high of 59 as determined by various methods and data. It was judged that ten independent agencies would develop custom mechanisms for five launches per mechanism. (Historically, multiple usage of one development has been the exception; individual developments have been more the rule.) The MMSE cost was estimated at $20M for two sets. However, individual developments would need only one set plus spare components and would be somewhat simpler on the average. An average custom development cost of only $10M was assumed. Installation, integration and maintenance costs were judged comparable for custom and MMSE mechanisms.

The RMS spin mechanism costing was based upon similar rationale. Basically, four custom developments could occur for TAD's, AMP's, and the gravity and magnetic measurement satellites. The TAD's deployment could be relatively simple while one of the higher spin rate S/C would require something comparable

to the RMS MMSE device. It was assumed that average development costs would be one-half the MMSE RMS device. The above is summarized below:

Bay Platform Sum - MMSE		
Design, fab (2 sets)	$ 20 M	
Bay Platform Sum - Custom		
Design, fab (1 set ea.) $10M x 10	100 M	
MMSE Savings Over No MMSE		$ 80M
RMS Sum - MMSE		
Design, fab (2 units)	$ 1.5M	
Other Equiv. Sum		
Design, fab (1 set ea.) 4 x $0.75M	$ 2.8M	
MMSE Savings Over Non-MMSE		$ 1.3M

Orbiter Spin Concept Savings. The savings potential over Thor-Delta launches by using the Orbiter's multiple/mixed payload capability to launch Thor-Delta class payloads is large since up to 4 of the largest such payloads (or more smaller payloads) can share a flight. These savings are not due uniquely to MMSE, but it is felt desirable to list the savings here for information purposes.

Costs for a Thor-Delta launch by an unsubsidized User has been quoted at $12.9M in 1976 dollars. $10M per launch was assumed since some smaller payloads will use smaller Thor-Delta configurations. Costs as low as $7+ M have been reported but prorated overhead was apparently not included. On the basis of $10.5M direct costs per Orbiter flight (without a TUG/IUS), similar payloads can be launched for about 1/4 of the Thor-Delta costs. However, 160 percent of the direct costs has been used to determine total Orbiter launch costs, including prorated overhead. Therefore, the latter cost comparison was also accomplished.

As discussed previously, a total of 50 Thor-Delta class payload launches is assumed. The only significant new equipment needed to adapt Thor-Delta type payloads to Orbiter deployment is the spin-up mechanism. This hardware is estimated to cost from $20M to $80M depending upon whether it is MMSE or custom (see preceding discussion of spin mechanism savings).

The results of the above considerations are summarized as follows:

Thor Delta Launch Costs

$10 M x 50 launches =	$ 500M

Orbiter Launch Costs

Direct launch costs, $10.5M x 50/4 =	$ 131.5M
Direct + Indirect launch costs, $10.5M x 1.6 x 50/4 =	$ 210 M
New Hardware (Spin Mechanism)	
- MMSE	$ 20 M
- Non-MMSE, 10 x $10M ea =	100 M

Orbiter Launch Savings Over Thor-Delta

Direct (only) Orbiter Costs:	MMSE Spin Concept	348.5M
	Non-MMSE Spin Concept	268.5M
Direct + Indirect Orbiter Costs:	MMSE Spin Concept	270. M
	Non-MMSE Spin Concept	190. M

Combined Savings

Figure 2-42 combines the savings from the above individual evaluation summaries. The total savings is seen to be quite substantial, even in the event of disagreement on some judgements and assumptions. Overall, it is felt that the error is on the conservative side, i.e., total 10 year savings are likely to be greater.

MMSE ITEM	COMPARED TO	FLIGHTS (OVER 10 YEARS)	TOTAL COST SAVINGS, $M
PMDM	CUSTOM HARDWIRE	500	75
STAR TRACKER	OTHER MMSE	64	2.6
ELECTRICAL CABLING	CUSTOM	500	17
FLUID LINES	CUSTOM	500	23
SPIN-UP MECHANISMS			
BAY PLATFORM SUM	SEMI-CUSTOM	50	80
RMS END EFFECTOR SUM	SEMI-CUSTOM	14	1.3
MAPPS	APPS	80 (+)	8
TOTAL MMSE ITEMS			207
STS MULTIPLE LAUNCH	THOR-DELTA LAUNCH	50	190-269*
OVERALL SAVINGS USING MMSE			397-476

*Orbiter Costs at $16.8M-$10.5M per Launch

Figure 2-42. Preliminary STS/MMSE Cost Savings Estimate

Note that the total savings number in Figure 2-42 could imply that the PMDM and electrical cabling savings are claimed simultaneously. In reality, if the PMDM concept is implemented and utilized, the necessary hardwiring would decrease to only that necessary for high data rate and caution and warning wiring. The hardwire costs (both MMSE and custom) would be expected to decrease, the effect being to reduce the net total savings. However, for the purpose of this evaluation and the degree of accuracy involved, the effect is not significant. The main purpose is to show that very substantial potential savings can occur. A refined estimate requires indepth study not possible on this contract.

Follow-On Study Tasks

Table 2-29 summarizes what has been judged the most pressing areas or promising items that should be funded for brief special emphasis study efforts is an immediate follow-on. Effort is envisioned at the $15-20K level each.

Table 2-29. Candidate List of Special Emphasis Subjects for Follow-on Study

NO.	SUBJECT	REASONS
1.	LARGE PAYLOAD (THOR-DELTA) SPIN-UP MECHANISM	LARGE POTENTIAL FOR USE IN EARLY STS ERA; PRESENT STUDY LIMITED IN SCOPE, NEED FURTHER CONCEPTUAL STUDY
2.	RTG COOLING KIT ENCAPSULATOR CONCEPTS	RTG ENCAPSULATOR PORTION OF SYSTEM NOT PREVIOUSLY STUDIED; NEED CONCEPTUAL STUDY TO BRING UP TO STATUS OF HEAT TRANSFER PORTIONS OF OVERALL SYSTEM PRIOR TO NEXT STUDY DEFINITION PHASE
3.	ORBITER/SPACELAB SYSTEMS SIMULATOR	PRELIMINARY CONCEPTUAL EXPLORATION STUDY NEEDED TO CHECK CONCEPT FEASIBILITY AND MAKE NEXT DEFINITION STUDY EFFECTIVE
4.	DESIGN IMPLEMENTATION CONCEPT FOR PAYLOAD MDM	NEED TO DETERMINE CONCEPTS FOR IMPLEMENTING PMDM INSTALLATION, CHANGES/IMPACTS TO ORBITER AND PAYLOAD WIRING AND INTERFACES
5.	RMS SPIN-UP STABILITY AUGMENTATION	IDENTIFY CONCEPTS FOR STABILIZING RMS TO HANDLE HEAVIER PAYLOADS, EXPLORE POINTING PROVISIONS
6.	NEW MMSE REQUIREMENTS PRELIMINARY EVALUATION	EVALUATE STATUS OF FLOATING PALLET, VARIABLE VOLTAGE POWER CONDITIONER, EMI DETECTOR, AND PAD COOLING FOR PAYLOAD AS POTENTIAL MMSE REQUIREMENTS
7.	FUTURE MMSE POSSIBILITIES STUDY	CONDUCT BRIEF LOOK AT NEXT GENERATION OPERATIONS (SPS) TO PERCEIVE POTENTIAL MMSE REQUIREMENTS AND COMMON EQUIPMENT

Item 1 is to further explore the basic approaches and concepts already identified in this study before embarking on a detailed definition study.

Item 2, RTG Encapsulator Concepts, is concerned with the encapsulator or cooling muff configuration for receiving heat by radiation/conduction from the actual unit on the spacecraft. The muff must be variable in its bay location yet must be retractable to permit spacecraft deployment. Concepts for support, vibration isolation, retraction, stowage, and variable position interfaces have not been previously identified to the level of the other elements in the total kit concept, and this special emphasis item will correct that imbalance.

Item 3, Orbiter/Spacelab Systems Simulator, is to briefly examine the concept and the possible approaches in order to narrow the possibilities to be defined in a subsequent definition study. The Orbiter/Spacelab simulator is intended to serve as a test instrument for payload/experiment developers to duplicate Orbiter/Spacelab electrical interfaces and support service characteristics (e.g., computer, data bus, controls and displays, caution and warning). Lower user costs and less risk of interface incompatibilities are the gains from use of the device.

Item 4 is intended to briefly examine the impacts on both Orbiter and payloads resulting from implementation of the PMDM concept. A brief look at the effect on signals, wiring interfaces, installation complexity, payload development cycle, and integration aspects is envisioned to improve the baseline of data available prior to intensive definition study of the PMDM itself.

Item 5, RMS Spin-Up Stability Augmentation, is to briefly explore simple means of adding stability in kit form so that RMS spin-up may be extended into higher regimes of payload masses and rotation rates.

The last two subjects, Items 6 and 7, are intended to continue along the lines of this study; to briefly look at a number of items already identified as potential MMSE, but not yet studied due to lack of funding, and to identify new MMSE candidates with promise of further cost savings.

Study Plans

At the Mid-Term it was agreed that detailed study plans would be provided for an additional four interesting items which need further study of such a magnitude that they would be beyond the scope of a "special emphasis" study item. As such, each would be a candidate for a separate conceptual definition study. These were chosen as follows:

1. Payload Station (PS) Controls and Displays
2. Orbiter/Spacelab Systems Simulator
3. Payload Integrated Pointing System
4. RTG Cooling Kit

The study plans generated for each of these four items are attached hereto as Appendix A5.

3. CONCLUSIONS

GENERAL

This preliminary analysis of the special emphasis items has clearly shown their conceptual feasibility. No development problems nor interference with Orbiter program is foreseen. At least two items will need hardware procurement go-ahead in FY '77, and four of the items require additional definition study in FY'76. The cost-saving potential through NASA implementation of these STS/MMSE items described herein is believed to be very substantial and merits serious consideration. Further study to cover additional, possibly equally attractive items appears warranted.

SPECIFIC CONCLUSIONS

1. Payload MDM - Simplified payload version of Orbiter MDM (PMDM) is a cost-effective way to minimize payload-Orbiter avionic interface complexity and resulting costs, requires FY'76 conceptual refinement study to support hardware procurement in FY'77.

2. Modified Orbiter Star Tracker - The MOST is well suited to use as a payload-pointing sensor, needs refined definition study in FY'76 leading to selection of best options for procurement in FY'78, and promises reduced operations costs relative to a previously recommended MMSE gimbaled tracker.

3. Multi-discipline Auxiliary Payload Power System - The concept provides "extra" power for several payload missions and is thus "MMSE"; it promises greater mission flexibility including more payload bay volume for productive payloads and considerably lower weight and cost than if provided individually by payloads or by the mission-unique APPS concept. Refined definition study is needed in FY'76 to prepare for development and flight use in 1980.

4. Payload Spin-up Mechanisms - The spin-up of small, low rpm satellites by means of a device on the Orbiter RMS (manipulator arm) appears feasible and could possibly be extended into higher mass/rpm ranges. Refined analysis is needed in FY'76 as RMS stiffness model data becomes available. The spin-up of multiple payloads of the Thor-Delta class appears feasible and shows promise of very large cost savings compared to the alternative Thor-Delta individual launches. Further definition of the need for and the mechanisms and trades involved for handling sequentially four or more payloads (spacecraft) is needed in FY'76 leading to hardware procurement in late FY'77.

5. Payload-Orbiter Electric Cables - The lack of availability of payload data in sufficient detail prevents a rigorous solution to minimize the number of cables required for all possible payloads. However, the analysis shows that two different common cabling approaches may be feasible and will require additional study when required data is obtained at a future time. Recent direction to Orbiter program for supplying these cables on the Spacelab will serve to crystalize these approaches which then in turn may be largely applicable for other payloads.

6. Payload-Orbiter Fluid Lines - Again, detailed payload fluid requirements data is not yet available. However, the general concept or approach of supplying standard equal (or non-equal) lengths of hard lines within compatible fluid groups plus standard lengths of flexible lines shows promise of greatly simplifying development, storage, handling, and installation with a net saving in cost compared to the "scramble" system wherein each payload is responsible for these connecting lines.

7. This study in addition to other studies at Rockwell indicates the desirability of placing the payload interface connections for fluids and electrical wiring at the payload rather than at various points in the Orbiter; as presently specified, payloads must provide their own connections to Orbiter and the routing complexities make this difficult for them. Recent direction to Orbiter program to provide these connections to the Spacelab interfaces resolves this problem for about 40 percent of the planned NASA missions which utilize Spacelab; this philosophy should be extended to <u>all</u> payloads in the interest of facilitating integration and potential commonality savings.

8. Follow-on Study Effort - Seven items briefly described herein are worthy of consideration as candidate special emphasis study items funded at $15-20K level each.

9. FY'76 Studies - Four of the special emphasis study items/concepts appear to provide distinct cost or other advantages if implemented as STS/MMSE, and these should be funded for further conceptual refinement/preliminary definition in FY'76.

10. FY'77 Hardware Starts - Two of the six special emphasis tasks appear to require hardware go-ahead in FY'77:

    a. Payload MDM
    b. Satellite In-bay Spin-up Mechanism

4.0 RECOMMENDATIONS

It is recommended that:

1. NASA plan for the development of the six cost-effective STS/MMSE items described herein.

2. NASA fund additional seven special emphasis study items as potential STS/MMSE to ascertain their feasibility and indicated cost effectiveness relative to alternatives

3. NASA fund additional conceptual/programmatic studies for individual items in accordance with the four study plans provided in this report (Appendix A5) for potentially attractive STS/MMSE items.

4. NASA provide funding in FY '76 for conceptual refinement studies of four STS/MMSE concepts already determined to be feasible and shown to be desirable for early flight operations.

5. NASA consider placing the payload interface for all electrical and fluid connections at the payload rather than at various designated Orbiter points to facilitate integration and commonality, as was done recently for Spacelab.

6. NASA seriously consider developing Shuttle multiple launch capability for spinning stages/spacecraft for future mission capture analyses as a major cost saving.

APPENDIX A1

LIST OF REFERENCES

1. Multi-Use Mission Support Equipment (NAS8-30847), April 1975.

2. Concept Design of the Payload Handling Manipulator System (JSC-09709), June 1975.

3. Requirements/Definitions Document Remote Manipulator System (JSC-10633), October, 1975.

4. Spinning Solid Upper Stage/Shuttle Integration Study (NAS9-14000), July 1975.

5. Utility of STS to Space Communication Community (NAS8-31435), October 1975.

6. Summarized NASA Payload Descriptions (from MSFC), July 1975.

7. Delta Spacecraft Design Restraints, DAC-61687, Revised 1973.

8. Burner II Interim Upper Stage Study, Orbiter Interface Definition, D180-18425-2 Rev. B, June 1975.

9. Manned Orbital System Concepts Study, MSFC, NAS8-31014, September 1975.

10. Mid-Term Briefing, SD 75-SA-0112, dated 5 September 1975, and Mid-Term Technical Progress Report, SD 75-SA-0111, dated 15 August 1975.

11. Study Plan, "Study of STS Subsystems and Components for Multiuse Mission Support Equipment", SD 75-SA-0013-1, 29 February 1975.

APPENDIX A2

MMSE REQUIREMENTS/STS APPLICABILITY SUMMARY

This Appendix contains the table which lists the outputs of Tasks 1 and 2 of the study. The table is divided into two sections, airborne and ground. The left-hand portion of the table (through "Requirements") is the Task 1 output and the remaining portion is the Task 2 output.

MMSE Airborne Requirements/STS Applicability Summary

SD ID	MMC MMSE Reference ID	MMC MMSE Reference Vol II Page No.	MMC MMSE Reference Vol III Page No.	Requirement	Potential STS/Other Applicability As Is	Potential STS/Other Applicability Mod	Potential STS/Other Applicability No	Disposition Rationale and Comments
				1.3.1 Structures (ST)				
				1.3.2 Propulsion (PR)				
				1.3.3 Power				
				1.3.3.1 Electrical Power Generation (EP)				
EP-R1		64	9, 39	* Auxiliary Power Unit (Tug and IUS)	X	-	-	Look at Low Cost Battery or batteries from other potential IUS. Additional batteries can be added to the IUS power system to satisfy MMSE requirement.
EP-R2	SD Requirement			(3) Auxiliary Power System	-	X		Large portion of system components assumed to come from Shuttle program
				1.3.4 Avionics				
				1.3.4.1 Guidance, Navigation & Control (GN&C)				
GN-R1		36	3, 33	*(2) Small IPS (Miniaturized Pointing Mount)	-	X	-	Goddard SIPS may be candidate (based on OSO-H)
GN-R2		36	4, 33	*(1) Boom Platform	-	-	X	Nothing existing in STS program - but something must exist such as autonavigator platforms
GN-R3		36	4, 36	* Inertial Measurement Unit (IMU)	-	-	X	Orbiter IMU drift rate is 35 times greater than MMSE requirement
GN-R4		36	4, 34	* Celestial Sensor - Coarse (10-30 arc-sec)	-	MC 431-0128	-	Shuttle tracker can be modified with lens substitution capability to give pointing accuracy up to 1 arc sec and 0.1 arc sec stability
GN-R5		36	4, 35	Celestial Sensor - Fine (0.5-1.0 arc-sec)	-	MC 431-0128	-	
GN-R6		36	6, 36	* Earth (Horizon) Sensor (180-360 arc-sec)	MC 432-0214	-	-	GPS item has sufficient accuracy to satisfy payload requirements
GN-R7		36	6, 37	(1) Solar Sensor (180 arc sec)	-	-	X	GPS sun sensors MC 432-0216 and MC 476-0143 are not accurate enough to satisfy this request
GN-R8		36	6, 37	Solar Sensor (0.5-1.0 arc sec)	-	-	X	GPS sun sensors MC 432-0216 and MC 476-0143 are not accurate enough to satisfy this request
GN-R9	SD Requirement			(3) Payload Integrated Pointing System	-	X	-	Portion of system components assumed to come from Shuttle program
				1.3.4.2 Communications and Tracking (CT)				
CT-R1		91	10, 42	* TV Camera (1024 lines)	-	-	X	No camera with this resolution in STS program and new camera will not be compatible with STS equipment. New design requires monitor which can switch to sweep rates compatible with 525 and 1024 lines.
CT-R2		91	10, 42	TV Camera (Commercial)	ICD 3-0050-01	-	-	Orbiter TV camera - 525 lines/frame can be used
CT-R3	Final Catalog IV-5			(2) Video Uplink System	-	-	X	Nothing as a system on Shuttle, individual pieces may be applicable
				1.3.4.3 Displays and Controls (D&C)				
DC-R1		82	10, 41	*(2) Payload Specialist Station (PSS)	-	X	-	Portion of system components assumed to come from Shuttle Program
				1.3.4.4 Instrumentation (IN)				
IN-R1		91	16, 42	* Protective Device - Earth/Moon/Sun Sensor	MC 431-0128	-	-	The protective device used in conjunction with the star tracker can be used
IN-R2		91	16, 42	* Protective Device - Radiation Detector	-	-	X	No suitable device on STS program
				1.3.4.5 Data Processing & Software (DP)				
DP-R1		69	10, 40	Mini/Micro Computer	X	-	-	There are no STS small (8000 word memory) computers available. A feasibility study of such a computer is underway at Autonetics (report due in July). The Autonetics A16 looks good.
DP-R2	SD Requirement			(3) Payload Multiplexer-Demultiplexer	-	MC 615-0004	-	MDM will be modified to provide a modular type unit to satisfy the various payload requirements
				1.3.4.6 Elect Power Dist & Control (PD)				
PD-R1		53	6, 38	* Regulator - 28 ± 1% Vdc	-	-	X	±2% is available, but no ±1%
PD-R2		53	6, 37	* DC/DC Converter - 5 Vdc	-	-	X	There is no separate DC/DC converter on Shuttle, GPS

MMSE Airborne Requirements/STS Applicability Summary (continued)

SD ID	MMC MMSE Reference			Requirement	Potential STS/ Other Applicability			Disposition Rationale and Comments
	ID	Vol II Page No.	Vol III Page No.		As Is	Mod	No	
				1.3.5 Environmental Control, Life Support				
				1.3.5.3 Thermal Control (TC)				
TC-R1		100	18, 48	* RTG Cooling Unit	-	X	-	RI MCR 1405 is being processed to design a kit capable of rejecting 50,000 BTU/hr with a total of 150,000 BTU - with doors open (kit is not used) orbiter must be pointed towards deep space for radiators to function adequately. Portion of system components assumed to come from Shuttle program.
				1.3.6 Crew Station and Equipment				
				1.3.6.2 Crew Provisions & Accommodations (CA)				
CA-R3	SD Requirement			(3) EVA Tool Kit	-	-	X	Nothing as total kit is currently part of Shuttle program
CA-R1		90	10, 42	* Payload Work Stations	X	-	-	Not MMSE - Orbiter has a general work table on lower aft deck. All stations identified by MMC will be permanently designed into orbiter or spacelab.
CA-R2	Final Catalog IV-7			(2) Manned Maneuvering Unit	-	-	X	Nothing in present inventory
				1.3.7 Mechanical Systems				
				1.3.7.7 Payload Bay Systems (PB)				
PB-R1		104	20, 48	* Orbiter/Payload Service Cables & J-Box Assy (Auto & IUS)	-	-	X	
PB-R2		112	23, 54	* Orbiter/Payload Service Lines & Conn. Plate Assy	-	-	X	
PB-R3		133	26, 98	Swing/Tilt Table	-	-	X	Only possibility might be large diameter Centaur IUS
PB-R4	1	137	26, 101	* Special Pallet	-	X	-	Spacelab pallet can be modified
PB-R5	2	137	26, 101	* Special Pallet	-	X	-	Spacelab pallet can be modified
PB-R6	GPP-1	141	26, 101	General Purpose Platform	-	-	X	DFI pallet may be acceptable as basic structure but cannot be used with tunnel
PB-R7	GPP-2	141	26, 101	General Purpose Platform	-	-	X	DFI pallet may be acceptable as basic structure but cannot be used with tunnel
PB-R8	PFM-1	143	26, 101	* Removable Pallet Floor	-	-	X	No similar hardware in STS program
PB-R9	PFM-2	143	26, 101	* Removable Pallet Floor	-	-	X	No similar hardware in STS program
PB-R10	1	147	26, 103	* Extendable Boom	-	-	X	No similar hardware in STS program
PB-R11	2	147	26, 103	* Extendable Boom	-	-	X	No similar hardware in STS program
PB-R12	3	147	26, 103	* Extendable Boom	-	-	X	No similar hardware in STS program
PB-R13	1	147	26, 103	* Erection/Deployment Mechanism	-	-	X	No similar hardware in STS program
PB-R14	2	147	26, 103	* Erection/Deployment Mechanism	-	-	X	No similar hardware in STS program
PB-R15	3	147	26, 103	* Erection/Deployment Mechanism	-	-	X	No similar hardware in STS program
PB-R16	MEM-1		26, 83	Module Exchange Mechanism	-	-	X	Goddard has a MEM as part of their Low Cost Modular Spacecraft program
PB-R17	SD Requirement			(3) Satellite Spin-Up Mechanism	-	-	X	Will be provided as one of the RMS end effectors.
PB-R18	SD Requirement			(3) RMS End Effectors	-	-	X	
				No WBS Category				
NC-R1		93	17, 44	* Purge System	-	A34-364	-	Modify this GSE item to mount in cargo bay or under the liner
NC-R2		93	18, 45	* Contam Monitor & Control (2) (Real Time Contam Monitor) (2) (Trace Contam Analyzer)	- - -	- - -	X X X	No integrated package exists but mass spec (C70-0879) and leak detector set (C70-0905) may do this job if packaged properly. See experiment ST-08.
NC-R3		93	18, 47	* Protective Shroud	-	-	X	Nothing similar on STS program
NC-R4		104	20, 52	* Payload Umbilical Assembly (IUS)	-	-	X	Shuttle has made no provisions for electrical cabling interfaces with payloads - Maybe carrier unique
NC-R5		104	23, 53	* Payload Service Cable (IUS)	-	-	X	
NC-R6		104	23, 53	* Payload Umbilical Assembly (Automated)	-	-	X	↓

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MMSE Airborne Requirements/STS Applicability Summary (continued)

SD ID	MMC MMSE Reference ID	Vol II Page No.	Vol III Page No.	Requirement	Potential STS/ Other Applicability As Is	Mod	No	Disposition Rationale and Comments
HC-R7		112	25, 57	* Connector Plate/PL Service Line Assy (IUS-Auto)	–	–	X	Shuttle has made no provisions for fluid connections with payloads
HC-R8		112	25, 57	* Connector Plate/PL Service Line & Pallet Inter. Conn Assy	–	–	X	
HC-R9		112	25, 57	* Payload Service Lines (IUS)	–	–	X	
HC-R10	PIA-1	120	26, 62	* Payload Interface Adapters – General Purpose	–	–	X	Nothing in present STS program regarding mechanical interface equipment between orbiter and payloads
HC-R11	PIA-2	120	26, 62	* Payload Interface Adapters – General Purpose	–	–	X	
HC-R12	PIA-3	120	26, 62	* Payload Interface Adapters – General Purpose	–	–	X	
HC-R13	PIA-4	120	26, 62	* Payload Interface Adapters – General Purpose	–	–	X	
HC-R14	PIA-5	120	26, 62	Payload Interface Adapters – Kick Stage (Tug)	–	–	X	
HC-R15	XPIA-1	120	26, 83	* Payload Interface Adapters – Kick Stage (IUS)	–	–	X	
HC-R16	PMB-1	120	26, 62	Payload Mounting Beam – Tug – Multiple P/L's	–	–	X	
HC-R17	PMB-2	120	26, 62	* Payload Mounting Beam – Tug – General Purpose	–	–	X	
HC-R18	PMB-3	120	26, 62	Payload Mounting Beam – Tug – Kick Stage	–	–	X	
HC-R19	XPMB-1	120	26, 83	* Payload Mounting Beam – IUS – Multiple P/L's	–	–	X	
HC-R20	XPMB-2	120	26, 83	* Payload Mounting Beam – IUS – General Purpose	–	–	X	
HC-R21	XPMB-3	120	26, 83	* Payload Mounting Beam – IUS – Kick Stage	–	–	X	
HC-R22	SLP-1	120	26, 74	* Separation Latch and Pushoff Assy	–	–	X	
HC-R23	PDK-1	120	26, 77	* Payload Docking Kit – Probe	–	Apollo	–	Manual mechanisms should be removed
HC-R24	PDK-2	120	26, 77	Payload Docking Kit – Drogue	Apollo	–	–	
HC-R25	PSP-1	120	26, 77	* Payload Services Plate – Actuated	–	–	X	
HC-R26	PSP-2	120	26, 77	* Payload Services Plate – Floating	–	–	X	
HC-R27	ITA-1	120	26, 77	IUS/Tug Adapter	–	–	X	
HC-R28	PLS-2	120	26, 78	Payload Spacer – Tug	–	–	X	
HC-R29	PLS-3	120	26, 78	Payload Spacer – Tug	–	–	X	
HC-R30	PLS-4	120	26, 78	Payload Spacer – Tug	–	–	X	
HC-R31	PLS-5	120	26, 78	Payload Spacer – Tug	–	–	X	
HC-R32	PLS-6	120	26, 78	Payload Spacer – Tug	–	–	X	
HC-R33	XPLS-2	120	26, 83	* Payload Spacer – IUS	–	–	X	
HC-R34	XPLS-3	120	26, 83	* Payload Spacer – IUS	–	–	X	
HC-R35	XPLS-4	120	26, 83	* Payload Spacer – IUS	–	–	X	
HC-R36	PHL-1	120	26, 83	Power Hinge & Latch Assembly	–	–	X	
HC-R37	SS-1	120	26, 83	Spin Separation Module	–	–	X	
HC-R38	PUS-1	133	26, 93	* Payload Umbilical Support	V70 340633	–	–	DFI pallet will satisfy the requirement but weighs 500 lb – Can be prototype if weight is a problem
HC-R39	APS-1	133	26, 93	Automated Payload Support	–	–	X	
HC-R40		147	26, 103	*(1) Airborne Canister	–	–	X	
HC-R41	Final Catalog III-9			(2) Large Payloads Transportation System	–	–	X	Super Guppy
HC-R42	SD Requirement			(3) Orbiter/Spacelab System Simulator	–	–	X	

*Items Required During 1st Two Years

(1) Deleted from final MMSE Catalog (6-75)

(2) Items identified by MMC final catalog (6-75) requiring FY '77 go ahead

(3) New SD requirements 6-17-75

MMSE Ground Requirements/STS Applicability Summary

SD ID	MMC MMSE Reference: ID	Init Cat. Page No.	Final Cat. Page No.	Requirement	Potential STS/Other Applicability: As Is	Mod	No	Disposition Rationale and Comments
				1.9.1 Ground Support Equipment				
				Auxiliary (A)				
A-R1	KHA-MI-44		4.27	Environmental conditioning unit	S70-0573	K311 S78-0108	–	S70-0707 (K600), S70-0708 (K601), all potential, S70-0573 may be replaced with Air Force F106 unit.
A-R2	KHB-MS-09		4.47	Payload purge cart	–	A34-364	–	A34-364 appears to satisfy requirements, S72-0694 (identified by MMC) is fixed equip on pad (distribution and control, service fluid umbilical, pressurize and purge GSE – 3000 psig) – S70-0534 (K599)
A-R3	KHC-MI-09		4.56	Portable horizontal clean room;	–	–	X	
A-R4	KHC-TH-04		4.61	TPF workstand clean room	–	–	X	Should be provided by TUG program as facility item.
				Checkout (C)				
C-R1	KHC-MS-10		4.62	Spacecraft buildup alignment set	–	–	X	
C-R2	KHC-MS-11		4.63	Spacecraft G&N alignment set	–	C70-0701	–	
C-R3	KHC-MS-12		4.64	Spacecraft electronics calibration set	–	C70-0716	–	C7-0721, 0727, and 0733 and others available but may need modification for calibration capability.
C-R4	KHC-MT-08		4.65	Cable sets	–	C72-1055	–	Some sets can be MMSE but others must be payload unique ≈ 20 cable sets exist now.
C-R5	KHC-MT-09		4.66	Breakout boxes set	C70-1087	–	–	
C-R6	KHC-MT-10		4.67	Ordnance circuits simulator	–	K653 C70-0534	–	
C-R7	KHC-MT-11		4.68	Payload electrical simulator	K39 C70-0547	–	–	
C-R8	KHC-MT-12		4.69	Comm and instrumentation test set	–	C70-0565	–	C70-0584 and C70-0646 may also do the job.
C-R9	KHC-MT-13		4.70	Spacecraft engine alignment test set	–	A70-0645	–	
C-R10	KHC-MT-14		4.71	Spacecraft G&N system test set	–	C70-0701	–	
C-R11	KHC-MT-15		4.72	Spacecraft power system test set	–	C70-0656	–	C70-0657 and C70-1050 may also do the job.
C-R12	KHC-MT-16		4.73	Spacecraft propulsion system test set	–	C70-0647	–	
				Handling (H)				
H-R1	KHA-MH-03		4.5	P/L container horizontal access equip	A70-0519	K1194 A70-0835	–	
H-R2	KHA-MH-06		4.7	S/C vertical assy stand access platform	–	–	X	This probably should be a facility item – see Item TH-45 (5.76)
H-R3	KHA-MH-10		4.9	Payload container	K934 A70-0806	–	–	
H-R4	KHA-MH-11		4.11	Payload element container	–	K934 A70-0806	–	Make smaller container from large one.
H-R5	KHA-MH-19		4.13	Payload handling fixture	K935 H70-0802	–	–	
H-R6	KHA-MH-27		4.17	Multipurpose sling set	H72-0736	H70-0506	–	H70-0504, etc., etc. (57 items in all) will also be applicable in part
H-R7	KHA-MH-29		4.19	Payload container sling set	K937 H70-0804	H78-3006 H78-3004	–	H70-0517, 0524, etc., etc. (8 items in all) will also be applicable in part
H-R8	KHA-MH-34		4.21	S/C vertical assembly stand	K939 A70-0808	–	–	
H-R9	KHA-MH-45		4.29	P/L container vertical access equip	–	–	X	This probably should be a facility item
H-R10	KHA-TH-55		4.31	Tug/payload handling fixture	H70-0802	–	–	Looks like payload handling fixture KHA-MH-19 (H70-0802) will satisfy this requirement
H-R11	KHA-MT-07		4.33	Payload mechanical simulator	K938 A70-0807	–	–	
H-R12	KHB-AH-30		4.37	P/L assembly/test horiz workstand	K1157 A70-0834	K942 A70-0811	–	A70-0810 (K941) may also do the job
H-R13	KHC-MH-20		4.58	Spacecraft rotation fixture	–	–	X	It appears as if this requirement could be satisfied by slings
				Packaging and Transport (P)				
P-R1	KHA-MH-26		4.15	Transportation instrumentation set	K1346 P78-3103	–	–	
P-R2	KHA-MH-39		4.23	P/L container horizontal transporter	K74 P70-0559	–	–	
P-R3	KHA-MH-41		4.25	P/L container vertical/element trans	K168 P77-0006	P70-0571	–	Redesign P70-0571 to transport equipment in horizontal orientation
P-R4	KHC-MH-12		4.57	PSS panel container	–	–	X	Could be payload unique – basic PSS will be mission kit.
P-R5	KHC-MH-46		4.59	Universal cover set	–	A70-0502	–	
P-R6	KHC-MH-47		4.60	Spacecraft storage cover	–	–	X	
				Servicing (S)				
P-R1	KHB-MS-01		4.39	Hydrazine service set	S70-0613	–	–	S70-0683 (K760) identified by MMC is too large, 2000 gal (≈ 1800 lb), mobile, 4 wheel tank trailer with controls
S-R2	KHB-MS-02		4.41	Instrument gas service set	–	C70-0743	–	Possible mods required for compatibility with 2e and pressure level (7 levels available on current item)
S-R3	KHB-MS-03		4.43	Liquid helium service set	–	–	X	S70-0695 (K325) – Fixed storage, distribution and control – orbiter safing, deservice, maint and C/O – 6000 psi gas
S-R4	KHB-MS-04		4.45	Liquid hydrazine service set	–	–	X	S70-0688 (K124) – 13000 gal, 5th wheel trailer – 40,000 lb gross
S-R5	KHB-SS-03		4.49	Liquid nitrogen service set	–	–	X	Nothing available but cart used for LOX or LH_2 would be applicable
S-R6	KHB-SS-04		4.51	Liquid neon service set	–	–	X	Nothing available but cart used for LOX or LH_2 would be applicable
S-R7	KHB-SS-05		4.53	Liquid oxygen service set	–	–	X	S70-0689 (K125) – 4000 gal, 5th wheel trailer – 40,000 lb gross

Should be MMSE not optional

Items identified by MMC final catalog (6-75) requiring FY'77 go-ahead.

MMSE Ground Requirements/STS Applicability Summary (KSC Payload Integration Office Responsibility)

SD ID	MMC MMSE Reference			Requirement	Potential STS/Other Applicability			Disposition Rationale and Comments
	ID	Init Cat. Page No.	Final Cat. Page No.		As Is	Mod	No.	
				1.9.1 <u>Ground Support Equipment</u>				
				<u>Auxiliary (A)</u>				
A-R5	MH-08		5.6	Desiccant breather	—	—	X	
A-R6	MH-21		5.10	Mobile electric power generator	—	—	X	
A-R7	MH-37		5.15	Payload tethered tool set	—	—	X	
A-R8	MS-08		5.21	Work light set	—	—	X	
A-R9	MT-04		5.26	Ground power supply set	—	—	X	
A-R10	MH-38		5.16	Standard inspection tools	—	—	X	
				<u>Checkout (C)</u>				
C-R13	MS-07		5.20	Payload battery charger/test set	—	—	X	
C-R14	MT-01		5.24	Hazardous gas detector	—	—	X	
C-R15	MT-03		5.25	Fluid system leak test set	—	—	X	
C-R16	MT-05		5.27	Ordnance device test set	—	—	X	
				<u>Handling (H)</u>				
H-R14	MH-01		5.5	General purpose access equipment	—	—	X	
H-R15	MH-14		5.7	Mobile crane	K270 H70-0692	—	—	U72-1138 may also do the job
H-R16	MH-17		5.9	Payload hoist fitting	—	—	X	
H-R17	MH-25		5.14	Ordnance handling set	—	—	X	
				<u>Packaging and Transport (P)</u>				
P-R7	MH-15		5.8	Lab cart/hand truck dolly	—	—	X	
P-R8	MH-22		5.11	Tie down kits	—	—	X	
P-R9	MH-23		5.12	Exterior prime mover	—	—	X	
P-R10	MH-24		5.13	Interior prime mover	K1112 P70-0802	—	—	P70-1000 (K-138) may also do the job
P-R11	MH-42		5.17	Component handling truck	P70-1000	—	—	
P-R12	MH-43		5.18	Flat bed trailer truck	—	—	X	
				<u>Servicing (S)</u>				
S-R8	MS-06		5.19	Nitrogen service set	—	—	X	
S-R9	SS-07		5.22	Oxygen service set	—	—	X	
S-R10	AS-02		5.23	Helium service set	—	—	X	

MMSE Ground Requirements/STS Applicability Summary (Incorporated in Another Item)

SD ID	MMC MMSE Reference ID	Init Cat. Page No.	Final Cat. Page No.	Requirement	Potential STS/ Other Applicability As Is	Mod	No	Requirement Covered by (Per MMC)	Disposition Rationale and Comments
				1.9.1 Ground Support Equipment					
				Auxiliary (A)					
				Checkout (C)					
C-R17	AS-07		5.42	Memory Load and Verify Unit	-	C70-0701	-	MMC-MT-14	
C-R18	TS-10		5.43	Spacecraft Memory Load and Verify Unit	-	C70-0701	-	MMC-MT-14	
C-R19	MT-06		5.44	PSS Simulator	K1157 A70-0834	K942 A70-0811	-	MMB-AH-30	A70-0810 (K941), may also do the job
C-R20	ST-05		5.45	Hardware Interface Modules (HIM)	-	-	X		
C-R21	AT-01		5.46	P/L Monitor, Test and Control Console	-	-	X		
C-R22	AT-10		5.47	P/L Antenna Test Hat Set	-	C70-0565	-	MMC-MT-12	C70-0584 and/or C70-0646 may also satisfy the requirement
C-R23	AT-11		5.48	Orbiter Mechanical/Elect (LIS) Simulator	-	-	X		
C-R24	TT-03		5.49	Hardware Interface Module (HIM)	-	-	X		
C-R25	TT-04		5.50	S/C Antenna Dummy Load Set	-	C70-0565	-	MMC-MT-12	C70-0584 and/or C70-0646 may also satisfy the requirement
C-R26	TT-06		5.51	S/C to Orbiter Interface Simulator	-	-	X		
C-R27	TT-18		5.52	S/C Pressurization System Test Set	-	-	X	MS-06	
				Handling (H)					
H-R18	MH-04		5.31	Kickstage Assy/Test Work Std Access Plat	-	-	X	MMA-MH-06	This probably should be a facility item
H-R19	MH-05		5.32	P/L Assy/Test Work Std Horiz Access Plat	K1157 A70-0834	K942 A70-0811	-	MMB-AH-30	A70-0810 (K941) may also do the job
H-R20	MH-31		5.33	Horizontal Spacecraft Sling Set	H72-0736 H70-0804	H70-0506	-	MMA-MH-27 MMA-MH-19	H70-0504, etc etc (57 items in all) will also be applicable in part
H-R21	MH-32		5.34	Spacecraft Shipping Container Sling Set	H72-0736	H70-0506	-	MMA-MH-27	H70-0504, etc etc (57 items in all) will also be applicable in part
H-R22	SH-04		5.36	Workstand Access Platform	K1157 A70-0834	K942 A70-0811	-	SH-13	A70-0810 (K941) may also do the job
H-R23	AH-16		5.37	P/L Assy/Test Stand Access Platform	K1157 A70-0834	K942 A70-0811	-	MMB-AH-30	A70-0810 (K941) may also do the job
H-R24	AH-29		5.38	Payload Strongback Sling Set	K937 H70-0804	H78-3004 H78-3006	-	MMA-MH-19	MMA-MH-29 is better match - H70-0517, 0524, etc (8 items in all) may also be applicable in part
H-R25	AH-33		5.39	S/C and P/L Container Tie Down Set	-	-	X	MMA-MH-10 MMA-MH-11	
H-R26	TH-26		5.40	TPF Workstand Access Platform	-	-	X	TH-45	
H-R27	TH-40		5.41	Payload Sling Set	H72-0736	H70-0506	-	MMA-MH-19	MMA-MH-27 is better match - H70-0504, etc (57 items in all) may also be applicable in part
				Packaging and Transport (P)					
P-R13	MH-40		5.35	Vertical Payload Container Transporter	K168 P77-0006	P70-0571	-	MMA-MH-41	Redesign P70-0571 to transport equipment in horizontal orientation
				Servicing (S)					

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MMSE Ground Requirements/STS Applicability Summary
(Provided by Another Source)

SD ID	MMC MMSE Reference			Requirement	Potential STS/ Other Applicability			To be Provided by (Per MMC)	Disposition Rationale and Comments
	ID	Init Cat. Page No.	Final Cat. Page No.		As Is	Mod	No		
				1.9.1 Ground Support Equipment					
				Auxiliary (A)					
A-R11	SH-18		5.69	Personnel Air Conditioning Unit	570-0573	K311 578-0108	-	Spacelab P.O.	Should be included in requirement A-R1 (KMA-MH-44)
A-R12	AH-38		5.73	P/L Bay Mobile Environ Conditioning Unit	570-0573	K311 578-0108	-	Shuttle Item	Should be included in requirement A-R1 (KMA-MH-44)
A-R13	TS-12		5.78	RTG Cooling Unit		X	-	Payload P.O.	If ASE unit is MMSE then GSE is also MMSE - May be common unit
A-R14	ST-04		5.81	Ground Power Supply Set	C70-0656	C14-0262	-	Spacelab P.O.	C70-0657, C70-0693 and 15 other units may be applicable
				Checkout (C)					
C-R28	TS-01		5.77	Standard Lab Calibration Equip	-	-	X	Facility	
C-R29	ST-03		5.80	Experiment Simulator	-	-	X	Spacelab P.O.	
C-R30	ST-07		5.82	Launch Processing System	-	-	X	Launch Site Facility	
C-R31	AT-03		5.83	P/L Bay Environ. Monitor and Control Equip.	-	-	X	Shuttle	Should be supplied as a part of the facility
C-R32	AT-15		5.84	Launch Processing System	-	-	X	Launch Site Facility	
C-R33	TT-08		5.85	Spacecraft to IUS Interface Simulator	-	-	X	Tug P.O.	
C-R34	TT-11		5.86	Launch Processing System	-	-	X	Launch Site Facility	
C-R35	MT-02		5.79	Latch Mechanism Test Kit	-	-	X	Tug P.O.	If Tug/PL adapters are MMSE, these test kits should also be MMSE
				Handling (H)					
H-R28	MH-02		5.55	P/L Bay Horizontal Access Equip	K14 A70-0519	-	-	Shuttle P.O.	This item was identified by MMC
H-R29	MH-18		5.58	P/L Adapter Element Handling Fixture	-	-	X	Flt MMSE P.O.	
H-R30	MH-28		5.59	P/L Adapter Elements Sling Set	H72-0736	H70-0506	-	Flt MMSE P.O.	Should be covered by H-R6 (KMA-MH-27)
H-R31	MH-30		5.60	Spacecraft, Vertical, Sling Set	H72-0736	H70-0506	-	Tug P.O.	Should be covered by H-R6 (KMA-MH-27)
H-R32	MH-33		5.61	P/L Adapter Element Stand	-	-	X	Flt MMSE P.O.	Tug Swing/Tilt Table may satisfy this requirement (ASE PB-R3)
H-R33	MH-35		5.62	Kickstage Assy/Test Work Stand	K939 A70-0808	-	-	Tug P.O.	May be covered by H-R8 (KMA-MH-34)
H-R34	SH-01		5.63	General Purpose Access Ladder	K30 A72-1013	-	-	S/L P.O.	A70-0562 (K29), A70-0514 (K231) may also satisfy the requirement
H-R35	SH-05		5.64	Bridge Crane	-	-	X	Facility	
H-R36	SH-07		5.65	Pallet Restraint Fixture	K935 H70-0802	-	-	S/L P.O.	Could be covered by H-R5 (KMA-MH-19)
H-R37	SH-10		5.66	S/C Comp/Elements Support Stand	-	-	X	S/L P.O.	
H-R38	SH-13		5.67	P/L Work/Assembly Stand	K1157 A70-0834	-	-	S/L P.O.	Should be covered by H-R12 (KMB-AH-30)
H-R39	SH-16		5.68	S/L Components Sling Set	H72-0736	H70-0506	-	S/L P.O.	Should be covered by H-R6 (KMA-MH-27)
H-R40	AH-07		5.70	Building Cranes and Hoists	-	-	X	Facility	
H-R41	AH-12		5.71	P/L Manipulator Handling Fixture	-	-	X	PCR	
H-R42	AH-18		5.72	P/L Bay Vertical Access Platform	-	-	X	PCR	
H-R43	TH-12		5.74	Building Cranes	-	-	X	Facility	
H-R44	TH-24		5.75	Cargo Bay Vertical Platform Set	-	-	X	PCR	
H-R45	TH-45		5.76	TPF Vertical Work Stand	-	-	X	Facility	
				Packaging and Transport (P)					
P-R14	MH-13		5.56	Adapter Protective Cover	-	-	X	Flt MMSE P.O.	If the Spacecraft storage cover KMC-MH-47 (P-R6) will not satisfy "As Is" it can be modified
P-R15	MH-16		5.57	Payload Component/Adapter Dolly	K224 P77-0018	-	-	Flt MMSE P.O.	

APPENDIX A3

MMSE EQUIPMENT SUMMARY AND MMSE ITEM DESCRIPTION SHEETS

This Appendix contains the outputs of Task 3 of the study and includes the MMSE equipment summaries (airborne and ground) and the item description sheets (62) for the Orbiter or other STS equipment which has been found to satisfy one or more of the MMSE requirements.

(Pages A3-2 and A3-3 are intentionally left out)

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE	- Airborne
ID NO.	- CA-H1
REQMT.	- CA-R1
ORIG.	- See Remarks

NAME: Payload Work Stations

PURPOSE: To provide work areas for payload operations

DESCRIPTION: Work stations are defined as any location in Shuttle or Spacelab where a task or activity relating to payloads is performed. Work stations include Orbiter PSS, and Spacelab airlocks, viewports, console/workbench and special locations.

PHYSICAL/FUNCTIONAL CHARACTERISTICS: TBD

INTERFACES:

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT:

REMARKS: The basic orbiter plus planned mission kits include necessary work stations.

MMSE ITEM DESCRIPTION

TYPE - Airborne
ID NO. - CT-H2
REQMT. - CT-R2
ORIG. - ICD 3-0050-01

NAME: TV Camera (Commercial)

PURPOSE: To monitor instrument performance via the payload specialist station.

DESCRIPTION: A typical TV camera using 525 lines per frame will satisfy this requirement.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Resolution - 525 TV lines
Optics Size - 15.2 cm (6 in) dia x 35.6 cm (14 in) long
Electronics Size - 40.6 x 25.4 x 15.2 cm (16 x 10 x 6 in)
Optics Weight - 5.9 Kg (13 lbs)
Electronics Weight - 9.1 Kg (20 lbs)
Power - 16 watts average

INTERFACES: Payloay Interfaces - TBD
PSS Interface - TBD

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: None

REMARKS:

MMSE ITEM DESCRIPTION

TYPE - Airborne
ID NO. - DC-H1
REQMT. - DC-R1
ORIG. - New

NAME: Payload Specialist Station

PURPOSE: To provide a central location in the orbiter for payload monitor and control equipment, data and communications management and other support equipment.

DESCRIPTION: The kit will be designed to include the following equipment:

1. Payload Monitor and Control

   Signal I/O Interface
   Computer/Computer Interface
   Status Indicators
   Multifunction Interactive Video Display
   Meters and Digital Readouts
   Event Timer
   Toggle Switches
   Rotary Switches
   Potentiometers
   Hand Controller (Pointing and Tracking)

2. Support Systems

   Computer/Computer Interface
   Display Support Electronics
   Payload Data Management

3. Data Storage

4. Accommodations for Payload Unique Panels

PHYSICAL/FUNCTIONAL CHARACTERISTICS: TBD

INTERFACES: Mission specialist station; on-orbit station; payloads; orbiter computer and/or data buses; orbiter ground link communications I/O.

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: N/A

REMARKS: The PS will be standardized as much as possible for use by individual and mixed (including military) payloads, providing general purpose computer support for display and control and other purposes, and interfacing with payloads via hardwire and standard data buses. Capability to add payload-unique functions will be provided.

MMSE ITEM DESCRIPTION

TYPE - Airborne
ID NO. - DP-H1
REQMT. - DP-R1
ORIG. - Autonetics Micron 16

NAME: Mini/Micro Computer

PURPOSE: To provide computer capability for carry-on programs with minimum interface complexity with the Orbiter computer.

DESCRIPTION: The computer will be integrated with the payload and Orbiter computer before payload installation into the orbiter. The computer will offer a variety of options permitting a broad range of applications. The options include the following:

- o Choice of wordlength - 4, 8, or 16 bits
- o Choice of memory types
- o Choice of input/putput

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Size - 10.2 x 15.3 x 1.3 cm (4 x 6 x 0.5 in)
Volume - 196.6 cm^3 (12 in^3)
Weight - 0.23 Kg (0.5 lbs)
Power - 4 Watts
Memory - 8000 words

INTERFACES:

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: None

REMARKS:

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE - Airborne
ID NO. - DP-H2
REQMT. - DP-R2
ORIG. - MC615-0004

NAME: Payload Multiplexer Demultiplexer (PMDM)

PURPOSE: To reduce interface signal cabling size and associated noise, installation, calibration, and interface circuit problems by providing bi-directional multiplexing and demultiplexing service between applicable payload interfaces for relatively low information rate/duty cycle signals.

DESCRIPTION: The PMDM converts and formats serial digital, analog, and discrete inputs from subsystems to a single wire, Manchester II coded data bus and decodes similar information received from the same data bus and outputs it to subsystems in serial digital, analog, or discrete format. The bi-directional information rate on the data bus can be up to 500,000 bits/sec (approximately). A separate computer provides addressing of the PMDMS on a given data bus. A PMDM consists of a case, mother board and connector assembly into which core modules (power supply, sequence control unit, analog-to-digital and digital-to-analog converters, and Manchester interface module adapter) and input-output modules plug. Eight input-output modules of any mix of available types can be used in a given PMDM to match subsystem/Payload interface requirements.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Size - 33 x 25.4 x 17.8 cm (13 x 10 x 7 in)
Weight - 8.3 kg (18.4 lbs) (with all 8 IOM's at 0.6 kg (1.3 lbs each)
Power - 53 watts peak

Subsystem inputs or outputs - Discretes - 0 to 5 or 0 to 28 volts
Analog - +5.12 volts to -5.12 volts (single ended or differential input)
Data bus serial digital - NRZ format
Interface: Data code - Manchester II (Bi Ø - L)
Data rate - 1 megabit/sec
Analog encoding accuracy: ± 0.5%

INTERFACES: Hardwire to subsystems/payloads (input and output)
Coaxial cable (one) data bus (input and output)
28 VDC (nominal) power
Computer/processor addresses all PMDMS on a data bus via the data bus.

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: Remove the redundancy configuration of the orbiter MDM by "splitting" the case in half so that it contains one set of core modules and space for 8 10M's. Remaining operation and 10M designs are unchanged.

REMARKS: Input-output modules as used for the solid rocket booster (SRB) may also be utilized in the PMDM without modification.

MMSE ITEM DESCRIPTION

TYPE - Airborne
ID NO. - EP-H1
REQMT. - EP-R1
ORIG. - IUS Battery

NAME: Auxiliary Power Generation

PURPOSE: To supply power and energy in excess of that provided by TUG or IUS.

DESCRIPTION: Payloads requiring more power than the 7 Kw available from the orbiter can obtain additional power from fuel cells and/or batteries. Additional fuel cells can provide 7 Kw of power per fuel cell. Batteries provide a flexible source of power where the weight and energy limitations are acceptable. Apollo/Skylab batteries used as modular units can provide from 1.2 to 12 Kw power with 1 to 4 batteries.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Typicsl Silver Zinc Battery - (Apollo/Skylab)
Rating - 500 Ampere Hours
Envelope - 47.7 x 34.8 x 42.6 cm (18.78 x 13.7 x 16.8 in)
Weight - 111.1 Kg (245 lbs)
Energy - 14.5 Kwh

INTERFACES: Orbiter electrical power system

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: None

REMARKS:

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE - Airborne
ID NO. - EP-H2
REQMT. - EP-R1
ORIG. - New

NAME: Auxiliary Power System

PURPOSE: To provide payload power and energy above that which can be supplied by the Orbiter.

DESCRIPTION: The kit includes one or two orbiter fuel cells mounted on special keel fittings between stations 1478.3 cm (582 in) and 1760.2 cm (693 in). All required electrical power distribution and conditioning equipment is also mounted off the same keel fittings. The fuel cells have the capability of 7 Kw each on a continuous basis and 12 Kw each for 15 minutes every 3 hours. Reactants for the fuel cells are stored in the Standard Orbiter mission extension kits, located underneath the payload bay liner. Fuel cell product water is routed to the Orbiter fuel cell product water tanks. Heat rejection is accomplished by a kit composed of one or two deployable radiator panels (duplicates of the Orbiter forward radiator panels - 27,000 BTU/Hr each) and an Orbiter flash evaporator (70,000 BTU/Hr). They are mounted on structure which can be attached to any Spacelab pallet, assuming adequate space is available [76.2 x 457.2 cm (30 x 180 in) section on outside diameter of payload envelope] or directly to a standard set of bridge fittings.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Weight - 1962.7 Kg (4327 lbs) (2 fuel cells and 2 cryo bits)
Size - Radiators - 76.2 x 457.2 cm (30 x 180 in)
Flash Evaporator - 50.8 x 50.8 x 76.2 cm (20 x 20 x 30 in)
Fuel Cells & Assoc. Equip - 279.4 x 152.4 x 38.1 cm (110 x 60 x 15 in)
Capacity - 1700 KwH
14 Kw continuous
24 Kw for 15 minutes every 3 hours

INTERFACES:

1. Orbiter fuel cell reactant storage
2. Orbiter fuel cell product water storage
3. Orbiter fuel cell vent and purge system
4. Orbiter electrical interface - Sta 1765.3 cm (695 in)
5. Payload heat exchanger
6. Payload electrical & heat transport connections.

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: N/A

REMARKS: The Orbiter designed fuel cells, radiator panels, flash evaporator, and power conditioning hardware will be used without modification. The Orbiter extended mission extension cryo kits and water tanks are used "in-place". Two Orbiter keel fittings will be redesigned to permit mounting of the fuel cells and associated equipment.

MMSE ITEM DESCRIPTION

TYPE - Airborne
ID No. - GN-H1
REQMT. - GN-R1
ORIG. - Goddard SIPS

NAME: Small IPS

PURPOSE: To fit within the spacelab airlock or to physically accommodate the instruments within the constraints of the Spacelab pallets and Orbiter cargo bay area.

DESCRIPTION: The small IPS is a small, three-axis stable platform which is compatible with the Spacelab airlock. Since the small IPS is a firm requirements for airlock operation, the performance of the small IPS is based on those payloads with instruments of 3 feet in diameter or less.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Diameter - 91.4 cm (3 ft) or less
Length - 457.2 cm (15 ft) or less
Weight - 498.9 Kg (1100 lbs) or less
Pointing Accuracy - 1 arc sec.
Pointing Stability - 1 arc sec.
Stability Rate - 0.0167 arc sec/sec
Gimbal Range - Hemispherical
Reference Sensors - Inertial, Solar, Earth

INTERFACES:

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: The small IPS is a derivative of the Inside-Out-Gimbal (IOG) proposed for Spacelab.

REMARKS:

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE	- Airborne
ID NO.	- GN-H4
REQMT.	- GN-R4
ORIG.	- MC431-0128

NAME: Star Tracker (10-30 arc sec)

PURPOSE: To accommodate payloads which require 10-30 arc sec pointing accuracy error signals

DESCRIPTION: Signals are provided to the orbiter flight control system or to MMSE IMU. This sensor is required for those payloads accommodated by shuttle systems. It is not required when Spacelab IPS is used.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Field of view - 6 degrees x 6 degrees
Star sensitivity - +6 magnitude star
Accuracy - 30 arcsec
Size of Star Tracker Envelope - 45.7 x 58.4 x 43.2 cm (18 x 23 x 17 in)
Size of Tracker Electronics - 40.6 x 30.5 x 12.7 cm (16 x 12 x 5 in)
Star Tracker Weight - 9.5 kg (21 lbs)
Star Tracker Electronics Weight - 8.2 kg (18 lbs)

INTERFACES:

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: Provide a 95 mm lens, a matching light shade and a bright object detector.

REMARKS:

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE - Airborne
ID NO. - GN-H5
REQMT. - GN-R5
ORIG. - MC 431-0128

NAME: Star Tracker (0.5 - 1.0 arc sec)

PURPOSE: To provide 0.5 to 1.0 arc sec pointing accuracy error signals for instruments without UV or white light optics.

DESCRIPTION: Provides signal to MMSE IMU or to instrument internal image motion compensation. Also provides signal to small IPS for instruments requiring better than 10 arc sec accuracy.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Field of view - 1.25 degrees x 1.25 degrees
Star sensitivity - +9 magnitude star
Accuracy - 0.8 arc sec
Star Tracker Optics Envelope - 45.7 dia x 152.4 cm long
(18 dia x 60 in long)
Star Tracker Electronics - 30.5 x 40.6 x 40.6 cm (12 x 16 x 16 in)
Star Tracker Electronics Weight - approximately 8.2 kg (18 lbs)

INTERFACES:

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: Provide a 456 mm lens, a matching light shade and a bright object detector.

REMARKS:

MMSE ITEM DESCRIPTION

TYPE - Airborne
ID NO. - GN-H6
REQMT. - GN-R6
ORIG. - MC432-0214 (GPS)

NAME: Earth Horizon Sensor (180 - 360 Arc Sec)

PURPOSE: To provide earth reference attitude error signals (180 arc sec) to Shuttle flight controls or MMSE IMU.

DESCRIPTION: This equipment is required for earth viewing payloads with pointing requirements better than 2 degrees provided by Orbiter. Spacelab provides an inertial reference of ± 1 arc sec, but no earth reference.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Envelope - 25.4 x 15.2 x 15.2 cm (10 x 6 x 6 in)
Volume - 5900 cu cm (360 cu in)
Weight - 7.7 Kg (17 lbs)
Pointing Accuracy - 180 arc sec

INTERFACES:

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: None

REMARKS:

MMSE ITEM DESCRIPTION

TYPE - Airborne
ID NO. - GN-H9
REQMT. - GN-R9
ORIG. - New

NAME: Payload Integrated Pointing System

PURPOSE: To provide a system which will satisfy a maximum range of pointing requirements.

DESCRIPTION: The Payload Integrated Pointing System consists of pointing sensors, IMU, and control electronics. The equipment will be mechanized so that a maximum number of payloads pointing requirements will be satisfied. The system will minimize the need for separate components and will optimize the control electronics so that a variety of dynamic requirements will be satisfied, i.e., size, accuracy, motion and attitudes.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

- o 3-axis reference - requiring 2 Trackers
- o Available stars dictated by platform orientation - +6 star sensitivity required
- o Continuous track

INTERFACES: TBD

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: N/A

REMARKS:

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE	- Airborne
ID NO.	- IN-H1
REQMT.	- IN-R1
ORIG.	- MC431-0128

NAME: Protective Device - Earth/Moon/Sun Sensor

PURPOSE: To indicate when instruments line-of-sight are pointed too close to earth, moon or sun.

DESCRIPTION: The protective device is a detector which senses radiation levels. This information is used to determine exposure durations and to indicate possible data degradation.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Envelope - 2.5 cm diam. x 7.5 cm length (1 in dia x 3 in long)
Volume - 85.2 cu cm (5.2 cu in.)
Weight - 0.5 kg (1 lb)
Sensor must detect the following limits:
Earth - 15, 20, 30, 50 degrees
Moon - 5, 45, 50 degrees
Sun - 15, 30, 45, 50, 90 degrees

INTERFACES: The protective device is used in conjunction with a star tracker.

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: None

REMARKS:

MMSE ITEM DESCRIPTION

TYPE - Airborne
ID NO. - NC-H23
REQMT. - NC-R23
ORIG. - Apollo

NAME: Payload Docking Kit - Probe

PURPOSE: Provide TUG with the capability to dock with payloads which require retrieval.

DESCRIPTION: This portion of Payload Docking Kit is a docking probe which provides radial clocking capability for alignment of the payload structural interfaces.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Envelope - 81.3 cm dia x 61 cm (32 in dia x 24 in)
Volume - 317.1 cu meters (11.2 cu ft)
Weight - 22.7 Kg (50 lbs)

INTERFACES: Interfaces with Payload Docking Kit - Drogue

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: Modifications to the Apollo probe include

- Redesign of probe assembly mounting provisions
- Incorporation of capability to radially "clock" the payload to provide proper alignment of the structure and services interface
- Changes to force/stroke curve of shock attenuators
- Removal of all provisions for manual operations

REMARKS:

MMSE ITEM DESCRIPTION

TYPE - Airborne
ID NO. - NC-H24
REQMT. - NC-R24
ORIG. - Apollo

NAME: Payload Docking Kit - Drogue

PURPOSE: Provide payloads which must be retrieved, with a TUG compatible docking interface.

DESCRIPTION: This portion of the Payload Docking Kit is a drogue which provides capture and axial and radial alignment of the payload with the interface equipment. Small radial and laterial misalignments are corrected by mating ramps on the payload interface adapters.

INTERFACES: Interfaces with Payload Docking Kit - Probe.

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: None

REMARKS:

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE - Airborne
ID NO. - NC-H38
REQMT. - NC-R38
ORIG. - V70-340633
(DFI PALLET)

NAME: Payload Umbilical Support

PURPOSE: To provide support for disconnectable/reconnectable fluid and electrical service connectors, and associated lines and bundles.

DESCRIPTION: The support is a welded aluminum tube structure; mounted to the payload bay hard points. Its primary functions are to support the moveable Payload Services Plate in the proper position to mate with the matching fixed plate on the payloads.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Envelope - 101.6 x 238.8 x 487.7 cm (40 x 94 x 192 in)
Volume - 3539.6 cubic meters (125 cu ft)
Weight - 401.4 kg (885 lbs)

INTERFACES: The support attaches directly to the orbiter payload bay hard points.

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: None

REMARKS:

MMSE ITEM DESCRIPTION

TYPE - Airborne
ID NO. - PB-H4
REQMT. - PB-R4
ORIG. - Spacelab

NAME: Special Pallet

PURPOSE: To provide a pallet which is similar to the Spacelab 2 segment pallet but with a 19000 lb payload capacity.

DESCRIPTION: The special pallet will accommodate payloads whose weights exceed the capacity of the Spacelab pallet (11,000 lbs). The special pallet will have the same dimensions as the 2 segment Spacelab pallet train, but with the payload capacity increased to 19,000 lbs.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Envelope - 271.8 x 487.7 x 599.4 cm (107 x 192 x 236 in)
Volume - 79287.2 cu meters (2800 ft^3)
Weight - TBD

INTERFACES:

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: Necessary structure changes to increase load-carrying capability.

REMARKS:

MMSE ITEM DESCRIPTION

TYPE - Airborne
ID NO. - PB-H5
REQMT. - PB-R5
ORIG. - Spacelab

NAME: Special Pallet

PURPOSE: To provide a pallet which is similar to the Spacelab 3 segment pallet but with a 16,500 lb capacity.

DESCRIPTION: The special pallet will accommodate payloads of up to 16,500 lbs. The special pallet will be dimensionally the same as the 3 segment Spacelab pallet which can accommodate 11,000 lb payloads.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Envelope - 271.8 x 487.7 x 599.4 cm (107 x 192 x 236 in)
Volume - 118930.8 cu meters (4200 cu ft)
Weight - TBD

INTERFACES:

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: Necessary structure changes to increase load-carrying capability.

REMARKS:

MMSE ITEM DESCRIPTION

TYPE - Airborne
ID NO. - TC-H1
REQMT. - TC-R1
ORIG. - New

NAME: RTG Cooling Kit

PURPOSE: To remove excess heat generated by payload RTG's.

DESCRIPTION: The kit will be capable of rejecting 50,000 BTU/Hr with a total capacity of 150,000 BTU. It will consist of a payload "muff", pump, supplementary GSE heat exchanger, flash evaporator, water storage tanks, lines and control valves. During operations, with the cargo bay doors closed, excess thermal energy is rejected by converting stored water to expendable steam. On-orbit with the doors open the heat will be rejected by the Orbiter radiators via the payload heat exchanger.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Size of heat exchanger - 25.4 x 25.4 x 43.2 cm (10 x 10 x 17 in)
Size of flash evaporator - 43.2 x 48.3 x 26.7 cm (17 x 19 x 10.5 in)
Water Tank - 40.6 cm dia (16 in) (2 required)
Accumulator Tank - 40.6 cm dia (16 in) (1 required)
Size of Controller - 25.4 x 25.4 x 27.94 cm (10 x 10 x 11 in)
Size of Freon Pump - 12.7 x 10.2 x 25.4 cm (5 x 4 x 10 in)

INTERFACES: Coolant lines interface with payload exchanger, fill and drain with the Orbiter fill and drain accommodations, coolant lines with the payload shroud and the steam vent with the Orbiter skin at a point to be determined.

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: N/A

REMARKS:

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE - Ground
ID NO. - A-H1
REQMT. - A-R1
ORIG. - S70-0707

NAME: Environmental Conditioning Unit

PURPOSE: To provide conditioned air to maintain payloads within environmental limits during transit in the payload and payload element containers.

DESCRIPTION: This unit will be towable from either end and contain the electrical generating system to power its air conditioning system. The air conditioning system will provide an air purge to the payload container at a flow rate of 0-90.7 kg/min (200 lb/min), temperature selectable within the range of 23.4° - 48.9°C (45 - 120°F), cleanliness nominally class 100, guaranteed class 5000 (HEPA filtered) air with 15 PPM or less hydrocarbons, and humidity equal to or less than 40%.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Weight - 2,449.4 kg (5400 lbs)
Fluid Media - Air
Pressure - 7.6 cm H_2O (3 in)
Capacity - NA

Flow Rate - 90.7 kg/min (200 lb/min) max.
Size - 152.4 x 182.9 x 243.8 cm (60 x 72 x 96 in)

INTERFACES: Payloads/payload containers

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: Provide means of generating required electrical power.

REMARKS: Two S70-0573 units are needed to supply 42.6 kg/min. (94 lbs/min). S78-0108 and S70-0708 can supply 90.7 kg/min (200 lbs/min) minimum. All three units also require modification to provide self-contained power generation system.

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE - Ground
ID NO. - A-H2
REQMT - A-R2
ORIG. - A34-364

NAME: Payload Purge Cart

PURPOSE: To provide a positive internal pressure to the payload to maintain internal cleanliness.

DESCRIPTION: The purge cart will be a mobile self-contained unit to supply small quantities of gaseous nitrogen or helium for internal purge and positive pressure. The unit will contain gas supplies; gauges, valves, regulators, hoses and fittings to interface with payloads or the payload container.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Weight - 272 Kg (600 lbs)
Fluid Media - Nitrogen
Pressure - 13789 Pascals (200 PSI)
Flow Rate - 5.6 m^3/Min (200 FT^3/Min)
Size - 1143 x 1117.6 x 914.4 min. (45 x 44 x 36 in)

INTERFACES: Requires 115 Vac ground power and 440 Vac if heater is desired.

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: Add racks for "K" bottles. Add tow bar.

REMARKS:

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE - Ground
ID NO. - A-H11
REQMT. - A-R11
ORIG. - S70-0708

NAME: Personnel Air Conditioning Unit

PURPOSE: To provide cooling air to the Spacelab module interior during all ground operations to maintain a habitable working environment.

DESCRIPTION: The unit will maintain 10,000 class cleanliness while simultaneously maintaining temperature and humidity levels in accordance with Federal Standard 209A. It will interface with the modules either through the crew transfer tunnel/orbiter interface location or through the aft bulkhead viewing port. It will interface with the modules only when installed in the work/assembly stand. It will provide the required temperature, humidity and cleanliness with the modules occupied by up to four persons.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Weight - 2449.4 kg (5400 lbs)
Fluid Media - Air
Pressure - 7.6 cm H_2O (3 in)

Flow Rate - 90.7 kg/min (200 lb/min) (min)
Size - 152.4 x 182.9 x 243.8 cm (60 x 72 x 96 in)

INTERFACES: Crew transfer tunnel or aft bulkhead
Electrical power supply

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: None

REMARKS: S70-0573 (2 units may be required), S70-0707 and S78-0108 are also applicable.

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE	- Ground
ID NO.	- A-H12
REQMT.	- A-R12
ORIG.	- S70-0707

NAME: Payload Bay Mobile Air Conditioning Unit

PURPOSE: To provide conditioned air for payload environment in the orbiter payload bay during transit from launch pad to OPF, OPF to VAB, and VAB to launch pad.

DESCRIPTION: This unit will be mobile, and contain the electrical generating system to power its air conditioning system. The air conditioning system will provide an air purge to the payload bay at a flow rate of 9-90.7 kg/min (200 lb/min.) temperature selectable within the range of 23.4° - 48.9°C (45-120°F) cleanliness nominally class 100, guaranteed class 5000 (Hepa Filtered) air with 15 PPM or less hydrocarbons, and humidity equal to or less than 40%.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Weight - 2449.4 kg (5400 lbs)	Flow Rate - 90.7 kg/min (200 lbs/min) (min)
Fluid Media - Air	Size - 152.4 x 182.9 x 243.8 cm (60 x 72 x 96 in)
Pressure - 7.6 cm H_2O (3 in)	

INTERFACES: Facilities, payload bay, payloads, GSE

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: Provide means of generating required electrical power.

REMARKS: Two S70 - 0573 units are needed to supply 42.6 kg/min (94 lbs/min) S70-0708 and S78-0108 can supply 90.7 kg/min (200 lbs/min) minimum. All three units also require modification to provide self contained power generation system.

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE	- Ground
ID NO.	- A-H13
REQMT.	- A-R13
ORIG.	- S70-0840

NAME: RTG Cooling Kit

PURPOSE: To remove heat generated in an RTG power unit during storage and ground operation following power supply installation and subsequent to orbiter landing.

DESCRIPTION: Water circulation unit, which can be mounted on orbiter post landing GSE Support transporter.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Weight - 2086.5 kg (4600 lbs.)
Fluid Media - Water
Pressure -
Operating Temp. - Out 23.4 - 48.9°C (45-120°F)
- In 65.6 - 107.2°C (150-225°F)
Flow Rate - 3.6 kg/min (8 lbs/min)
Size - 182.9 x 152.4 x 144.8 cm (6 x 5 x 4.75 ft)

INTERFACES: GSE heat-exchanger of airborne system

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: None

REMARKS:

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE - Ground
ID NO. - A-H14
REQMT. - A-R14
ORIG. - C70-0693

NAME: Ground Power Supply Set

PURPOSE: To provide power to the experiments which is normally provided in flight by the orbiter. Power shall be 24 to 32 VDC with a 4.0 kw capacity.

DESCRIPTION: DC power supply consists of patch panel distribution boxes, patchboards, patchcards, input/output connectors, terminal strips, overload protection circuitry, dummy loads, regulators, display devices and the appropriate wiring and cabling.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Weight - TBD
Electrical Output - 24 - 40 dc 500 amps
Power Regulation - ± 0.1%
Efficiency - 65% (min.)
Size - 119.4 x 61 x 50.8 cm (47 x 24 x 20 inches)

INTERFACES: 480 VAC, 60 Hz, 3 Phase Ground Power Supply

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: None

REMARKS: C14-262 can provide approximately 3 AMPS but includes checkout capability which is not required to satisfy the basic requirement. C70-0807 and C72-0810 may also satisfy the requirement.

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE - Ground
ID NO. - C-H2
REQMT. - C-R2
ORIG. - C70-0701

NAME: Spacecraft G & N Alignment Set

PURPOSE: To provide the capability to verify that a spacecraft is erect to local vertical and to test the null orientation of the star and sun sensor, gyros and momentum wheels.

DESCRIPTION: The unit shall include an Alignment Fixture, Targets, Mirrors, and Electronic Level.

PHYSICAL/FUNCTIONAL CHARACTERISTICS: TBD

INTERFACES: Spacecraft G & N test set, spacecraft, spacecraft and payload assembly/test stands, facilities.

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: Include null orientation, specific sun sensor capability, and gyro and momentum wheel functional verification.

REMARKS:

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE - Ground
ID NO. - C-H4
REQMT. - C-R4
ORIG. - C70-0519

NAME: Cable Sets

PURPOSE: To connect payload/payload elements to ground test and servicing equipment and ground power supplies, and to provide a grounding path.

DESCRIPTION: Cables and J-boxes to provide power and signal paths between the payloads and the checkout and servicing equipment.

PHYSICAL/FUNCTIONAL CHARACTERISTICS: TBD

INTERFACES: Payload elements; test and servicing equipment.

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: None

REMARKS: Cable sets C72-1049 and C72-1055 may also satisfy the requirement.

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE	- Ground
ID NO.	- C-H5
REQMT.	- C-R5
ORIG.	- C70-1087

NAME: Breakout Box Set

PURPOSE: To mate with payload element cables to monitor in-line signals and to input and probe connector pins and sockets for trouble shooting, test and verification.

DESCRIPTION: The set will consist of Apollo designed electrical breakout/breakthru boxes and interface adapter cables.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Weight - TBD
Size - 7.6 x 10.2 x 43.2 cm (3 x 4 x 17 in)

INTERFACES: Payload elements

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: None

REMARKS:

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE - Ground
ID NO. - C-H7
REQMT. - C-R7
ORIG. - C70-0547

NAME: Payload Electrical Simulator

PURPOSE: To provide a simulation of payload to PSS signals for PSS panel test, checkout and interface verification.

DESCRIPTIONS: The unit will provide and monitor signals to the Orbiter during checkout of the Orbiter. It will insure payload/Orbiter electrical, instrumentation and RF closed loop compatibility.

PHYSICAL/FUNCTIONAL CHARACTERISTICS: TBD

INTERFACES: PSS in Level 1 Integration Simulator

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: None

REMARKS:

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE - Ground
ID NO. - C-H8
REQMT. - C-R8
ORIG. - C70-0565

NAME: Communications and Instrumentation Test Set

PURPOSE: To activate payload communication and instrumentation systems transmit and receive data and commands via open loop or hard line, verify system operation, and record system output.

DESCRIPTION: The unit will have the capability of performing an end-to-end system test of the payload RF systems and associated LRU's via RF link. This includes S-band (SGLS, STDN/TDRS, P/L interrogator OFI/DFI transmitters, FM multiplexers and signal processors), UHF band (Transceivers), L-band (TACAN and ATC), C-band (beacon and radar altimeters), Ku-band (MSBLS and rendezvous radar) and RF antennas and switching. Downlink television and up/down audio and data links will be verified via RF link, RF interfaces with the vehicle antennas will be via hat couplers and re-radiating GSE antennas, stimuli and monitoring instrumentation will be mounted in a multi-bay console and will have the capability of manual/semi-automatic control. Additionally, the unit will have an ACE interface for transmission of blocks of up-command messages via RF link.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Weight - 2494.7 kg (5500 lbs)
Size - 182.9 x 670.6 x 91.4 cm (72 x 264 x 36 in)
Elect. Power - 120/208V, 60Hz 3Ø, 20 AMPS/Ø

INTERFACES: Spacecraft communications system

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: None

REMARKS:

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE - Ground
ID NO. - C-H9
REQMT. - C-R9
ORIG. - A70-0645

NAME: Spacecraft Engine Alignment Test Set

PURPOSE: To measure and verify proper alignment of spacecraft propulsion engines and attitude control thrusters relative to the spacecraft longitudinal axis.

DESCRIPTION: The set will consist of a fixture which attaches to the engine's thrust chambers exit and utilizes guide pins to align with alignment marks on the engine's thrust chambers.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Weight - TBD
Size - TBD

INTERFACES: Facility electrical power 110 VAC, 60 Hz, single phase.

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: Provide series of fixtures which are compatible with the spacecraft thrusters. (Main propulsion and attitude control).

REMARKS:

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE - Ground
ID NO. - C-H11
REQMT. - C-R11
ORIG. - C70-1050

NAME: Spacecraft Power Systems Test Set

PURPOSE: To energize spacecraft electrical systems and to check the spacecraft distribution network voltage levels.

DESCRIPTION: The unit will consist of a two bay console containing equipment which will provide AC/DC stimuli, apply loads to the outputs and measure voltage, current and display status of discrete outputs. The console will provide a work shelf and GSE adapter cables to interface with the subsystem being tested with the GSE interface panels. The GSE will provide over-current and over-voltage protection for inputs to the subsystem. The GSE will be cooled by self-contained blowers.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Weight - TBD
Size - TBD

INTERFACES: Facility 120/208 v, 60 Hz and 115/200 v 400 Hz power sources.

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: None

REMARKS: The C70-0547 also has the capability to satisfy the basic requirement.

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MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE	- Ground
ID NO.	- C-H12
REQMT.	- C-R12
ORIG.	- C70-0647

NAME: Spacecraft Propulsion System Test Set

PURPOSE: To verify spacecraft propulsion systems by sending typical system commands and verifying that responses are within success tolerances.

DESCRIPTION: The unit will have the following capabilities:

1. To record the signatures of the thruster valves on an oscillograph. (flight drivers required in GSE)
2. To perform channelization checks of the control and response circuits of the isolation and thruster valves.
3. Display instrumentation conditioned temp. and press. transducer outputs.
4. To check the solenoid driver pressure transducer feedback circuit.
5. To check the propellant servicing quantity (point sensor) gaging system.
6. To actuate and indicate the position of the electro-mechanical operated RCS doors.
7. To verify thruster burn through detector.
8. To verify valve arc suppression networks.
9. To interface with a remote firing system to permit engine static firing WSTF.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Weight - TBD
Size - TBD

INTERFACES: Spacecraft

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: Instrumentation may require modification to be compatible with the spacecraft systems.

REMARKS:

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE - Ground
ID NO. - C-H22
REQMT. - C-R22
ORIG. - C70-0565

NAME: Payload Antenna Test Hat Set

PURPOSE: Secure over payload antennas to contain RF radiation and route it via hardlines to test and check out equipment during spacecraft and kick stage test and checkout.

DESCRIPTION: This unit will have the capability of performing an end-to-end system test of payload RF systems. RF interface with the vehicle antennas will be via hat couplers and re-radiating GSE antennas. Stimuli and monitoring instrumentation will be mounted in a multi-bay console and will have the capability of manual/ semi-automatic control.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Weight - 2494.7 kg (5500 lbs)
Size - 182.9 x 670.6 x 91.4 cm (72 x 264 x 36 in)

INTERFACES: Facility power - 120/208 v, 60 Hz, 3 phase, 20 amps/phase

MODIFICATIONS TO ORIGINAL EQUIPMENT: None

REMARKS:

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE - Ground
ID NO. - C-H25
REQMT. - C-R25
ORIG. - C70-0565-1
C70-0565-2

NAME: Spacecraft Antenna Dummy Load Set

PURPOSE: To provide a set of dummy loads to match the impedance of the antenna under test and provide a check of the power capacity.

DESCRIPTION: The communications and tracking system test set will the following capabilities:

1. System test of operating modes and parameters associated with the S-Band, UHF, TACAN, MSBLS, Radar Altimeter, Ku Comm and Rendezvous Radar subsystems.
2. Receive RF and hardline down data, voice and video from the orbiter and condition for transmission to ground systems for data processing.
3. Receive voice and simulated payload data from ground systems, format and convert to RF carriers for transmission to the orbiter.
4. Generate single uplink commands, generate and interleave digital voice and transmit via hardline to orbiter signal processors or transmit via RF carriers to orbiter RF equipment.

PHYSICAL/FUNCTIONAL CHARACTERISTICS: TBD

INTERFACES:

Vehicle - The C70-0565 interfaces with the orbiter communications and tracking system indirectly, via other GSE, by one of the following means:

1. RF air link to all orbiter antennas via GSE antenna couplers and RF transmission lines.
2. RF hardline to on board RF equipment or transmission lines.
3. Hardline connection to the Network Signal Processors 1 and 2; the Payload Signal Processors 1 and 2 and the FM signal processor via GSE umbilicals and GSE cabling.

Facility -

1. Power - 120 VAC, 60 Hz, 3Ø
2. Cooling - Forced air cooling, TBD cu. ft./min., TBD °C through bottom access ducting. The maximum heat to be dissipated is estimated at 2000 BTU/hr. per console.

INTERFACES: (continued)

Facility - (cont.)

3. Supports - work benches or carts to support portable NAVAID test equipment and portable standard test equipment.

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: None

REMARKS:

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE	- Ground
ID NO.	- H-H3
REQMT.	- H-R3
ORIG.	- A70-0806

NAME: Payload Container

PURPOSE: To house all configurations of payloads during transfer from the various payload processing facilities to the orbiter OPF/pad and return.

DESCRIPTION: The container will be sized equal to the orbiter payload bay. Pickup points/retention fittings will be similar in type, quantity and location to the orbiter. Access doors will be along the top of the container and operate identical to the orbiter doors relative to allowable envelopes and clearances. Viewports will be provided and provisions for personnel access to the interior from ground level. Included are service panels, tie downs, and lift points to allow rotation of the loaded/unloaded container. Its closure device and external sizing will be compatible with the PCR. One end is hinged to allow vertical P/C installation.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Weight - 45359 kg (100,000 lbs) Size - 6.4 x 5.5 x 22.9 m (21 x 18 x 75 ft)

INTERFACES:

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: None

REMARKS:

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE - Ground
ID NO. - H-H4
REQMT. - H-R4
ORIG. - A70-0806

NAME: Payload Element Container

PURPOSE: To house spacecraft and Spacelab payloads (less than 25 feet in length) during transfers from building to building within the launch site.

DESCRIPTION: The container will be sized for a payload envelope of 15' dia x 25' long. Pickup points and retention fittings will be similar in type and location to those provided in the orbiter payload bay. Closure shall be identical to orbiter closure relative to allowable envelopes and clearance. The container shall have viewports to allow viewing of interior blindspots and shall be provided with personnel access to the interior from ground level. It shall contain interface service panels to allow interface with the appropriate ground power, environmental conditioning, environmental monitoring and RTG cooling systems. It shall be outfitted with appropriate tie-down and lifting capabilities.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Weight - 24267 kg (53,500 lbs) Size - 6.4 x 5.5 x 12.2 m (21 x 18 x 40 ft)

INTERFACES:

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT:
- Make shorter version of A70-0806
- Leave off side rails which were for use in PCR.

REMARKS:

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE - Ground
ID NO. - H-H5
REQMT. - H-R5
ORIG. - H70-0802

NAME: Payload Handling Fixture

PURPOSE: To support payloads in the horizontal and vertical positions and to provide a means of lifting payloads.

DESCRIPTION: The fixture (strong-back) will be a rigid frame device consisting of beams, cables, attach hook devices and rings adjustable to accommodate varying lengths and shifting C.g's of payloads up to 15' dia x 60' long and 65,000 lbs weight. It will interface with the payload on a non-interference basis such that engagement and load transference to attachment/retention points can occur while the handling fixture is still attached. It will support an IUS/TUG with payload by attachment to the carrier only, and automated by attachment to the spacecraft or to a spacecraft-orbiter adapter. It will not induce any bending or twisting loads on any payload element.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Weight - 15876 kg (35000 lbs) Size - 2.1 x 5.2 x 18.3 m
Capacity - Payloads 4.6 m (15 ft) (7 x 17 x 60 ft)
diameter and 18.3 m (60 ft) long

INTERFACES:

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: None

REMARKS:

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE - Ground
ID NO. - H-H6
REQMT. - H-R6
ORIG. - See (1)

NAME: Multipurpose Sling Set

PURPOSE: To provide general purpose lifting capability in conjunction with cranes or building hoists.

DESCRIPTION: The set will consist of a variety of spreader bars, hooks, clevises, drop cables and straps which will be used to lift items for which specific sling sets have not been designated. This would include such items as test or service sets, shipping containers, and spacecraft for which spacecraft contractors have not provided a special sling.

PHYSICAL/FUNCTIONAL CHARACTERISTICS: TBD

INTERFACES:

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT:
- Provide additional attach points on individual slings
- Accumulate various slings into multipurpose set

REMARKS:

(1) 57 sling sets now identified as GSE end items

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE - Ground
ID NO. - H-H7
REQMT. - H-R7
ORIG. - H70-0804

NAME: Payload Container Sling Set

PURPOSE: To lift loaded or unloaded payload and payload element containers in the horizontal attitude and for rotation of the payload containers.

DESCRIPTION: The sling set consists of two spreader bars and three cable assemblies complete with attachment fittings to interface with the payload or payload element containers. Two short cable assemblies will be used for lifting in the horizontal attitude, while a long and short will be used for erection to vertical.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Weight - ~2268 kg (5000 lbs)
Fluid Media - N/A
Pressure - N/A
Capacity -
Flow Rate - N/A
Size -

INTERFACES: Facility cranes and payloads.

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: None

REMARKS: Sling sets H78-3004, H78-3006 and H78-3007 would also be applicable if modified to provide the proper attach points.

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE - Ground
ID NO. - H-H8
REQMT. - H-R8
ORIG. - A70-0808

NAME: Spacecraft Assembly Vertical Stand

PURPOSE: To support automated spacecraft in the vertical orientation for final assembly and test prior to payload buildup. It must support multiple spacecraft with adapters for multispacecraft buildup and alignment prior to installation on the Tug/IUS.

DESCRIPTION: This stand will consist of a 4.6 m (15 ft) diameter rigid base plate with leveling legs on which are mounted 6 radial rails spaced 60 degrees apart running to the edge of the base plate. Payload interface fittings are mounted on the rails and slide radially to accommodate all spacecraft diameters.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Weight - TBD
Fluid Media - N/A
Pressure - N/A
Capacity - TBD
Flow Rate - N/A
Size - TBD

INTERFACES: Payloads, facilities

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: None

REMARKS:

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE - Ground
ID NO. - H-H10
REQMT. - H-R10
ORIG. - H70-0802

NAME: Tug/Payload Handling Fixture

PURPOSE: To be used in conjunction with a crane to lift the tug or IUS with payload during removal from TPF vertical workstand and installation into the payload container.

DESCRIPTION: The tug/payload handling fixture (strong back) will be a rigid frame device consisting of beams, cables, and attachment devices adjustable to accommodate various lengths and shifting C.G.'s of the tug or IUS with payload. It will interface with the tug/IUS only and will be capable of installing or removing the complete payload from the payload container in the vertical attitude.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Weight - 15876 kg 9(35000 lbs)
Fluid Media - N/A
Pressure - N/A
Capacity - TBD
Flow Rate - N/A
Size - 18.3 x 5.2 x 2.1 m
(60 x 17 x 7 ft)

INTERFACES: Payloads, payload cannisters, pads - P/L room

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: None

REMARKS:

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE - Ground
ID NO. - H-H11
REQMT. - H-R11
ORIG. - A70-0807

NAME: Payload Mechanical Simulator

PURPOSE: For payload facility and support equipment verification prior to processing a flight payload.

DESCRIPTION: This unit will consist of a 4.6 m (15 ft) diameter by 18.3 m (60 ft) long shell with a core tank approximately 15 feet in diameter. The core tank would be compartmentalized and could be filled with water as needed to adjust weight and center of gravity. The entire unit would be segmented in 3.0 m (10 ft), 6.1 m (20 ft) and 9.1 m (30 ft) lengths. Sliding pickup points would mount on rails to simulate Orbiter attach and lifting fixture interfaces at any desired location.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Weight - 29483 kg (65000 lbs)
Fluid Media - N/A
Pressure - N/A
Capacity - N/A
Flow Rate - N/A
Size - 18.3 m (60 ft) high, TBD wide, 4.6 m (15 feet) deep

INTERFACE: Orbiter - payload attach points, facility

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: None

REMARKS:

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE - Ground
ID NO. - H-H12
REQMT. - H-R12
ORIG. - A70-0810
A70-0811

NAME: Payload Assembly/Test Horizontal Work Stand

PURPOSE: To provide access and support to automated payloads for assembly, disassembly, and Level 1 interface simulation activities.

DESCRIPTION: The workstand is a horizontal support structure which supports and provides access to individual and integrated automated payloads. Retension fittings are similar in type, quantity and location to those provided in the Orbiter. Access is provided for the entire length of the workstand on both sides as well as the full width at both ends, for payload heights to 15 feet, and to all required positions within the envelope of the workstand. The stand includes cable trays for routing of electrical and fluid lines and interface panels for simulating the Orbiter to payload interface. Test equipment needed to perform Level 1 interface simulation is included.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Weight - 45359 kg (100,000 lbs)
Fluid Media - TBD
Pressure - TBD
Capacity - TBD
Flow Rate - TBD
Size - 3.7 x 6.4 x 18.3 m (12 x 21 x 60 ft)

INTERFACES: Payloads, facilities

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: Provide universal functional and physical interfaces.

REMARKS:

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE	- Ground
ID NO.	- H-H15
REQMT.	- H-R15
ORIG.	- H70-0692

NAME: Mobile Crane

PURPOSE: To provide the capability, when used with the appropriate handling fixtures and/or slings, to lift the various spacecraft or component shipping containers, as well as payload and payload element containers, and transfer these items from one position or device to another whenever this need exists outside of the processing facilities. Loads handled must include empty as well as fully loaded containers plus weight of handling fixture/sling.

DESCRIPTION: The unit consists of a 127005 kg (140 Ton) (Manitowoc Model 3900 T) Mobile Crane

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Weight - TBD
Fluid Media - N/A
Pressure - TBD
Capability - 127005 kg (140 ton)
Flow Rate - N/A
Size - TBD

INTERFACES: Payload attach points

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: None

REMARKS: U72-1138 - also has sufficient capacity to satisfy the requirement.

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE - Ground
ID NO. - H-H19
REQMT. - H-R19
ORIG. - A70-0810
A70-0811

NAME: Payload Assy/Test Horizontal Workstand Access Platform

PURPOSE: To provide personnel access to automated and spacelab payload elements when located in the horizontal payload assembly/test workstand and Level 1 interface simulator.

DESCRIPTION: The access platform is a horizontal structure which provides support and access to payloads for the entire length of the workstand on both sides as well as the full width at both ends for payloads up to 4.6 m (15 ft) in diameter and 18.3 m (60 ft) long. It also provides the means of accessing all positions within the envelope of the structure.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Weight - 45359 kg (100,000 lbs)
Fluid Media - N/A
Pressure - N/A
Capacity - 4.6 m (15 ft) diameter x 18.3 m (60 ft) long
Flow Rate - N/A
Size - 3.7 x 6.4 x 18.3 m (12 x 21 x 60 ft)

INTERFACES:

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT:
- Provide universal functional and physical interfaces
- Provide means of accessing all interior areas of workstand.

REMARKS:

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE - Ground
ID. NO. - H-H28
REQMT. - H-R28
ORIG. - A70-0519

NAME: Payload Bay Horizontal Access Equipment

PURPOSE: To provide personnel access to payload-Orbiter and payload-handling fixture locations when loading or removing an automated or spacelab payload from the Orbiter in the OPF.

DESCRIPTION: This set of ladders and platforms will provide access to all interior surfaces of the payload bay of horizontal orbiters, and to all systems and components installed in the bay. Access will be provided with the doors open or closed. In the latter case, ability to reach the entire door interior surfaces, systems, and radiators will also be provided. In the orbiter condition in which there is no payload or payload bay liner installed it will be compatible with the Horizontal Mid-Fuselage Equipment Access Set so that the combination of the two sets will provide complete access to the entire mid-fuselage interior. The set will be capable of being utilized for maintenance and repair operations of flight orbiters, and for the Static Test Article (STA).

PHYSICAL/FUNCTIONAL-CHARACTERISTICS:

Weight - TBD
Fluid Media - N/A
Pressure - N/A
Capacity - N/A

Flow Rate - N/A
Size - TBD

INTERFACES: Orbiter access platforms, orbiter payload bay.

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: Equipment must be capable of supporting the weight of at least two men simultaneously and shall be equipped with adequate safety features to preclude personnel harm.

REMARKS:

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE - Ground
ID NO. - H-H30
REQMT. - H-R30
ORIG. - H72-0736

NAME: Payload Adapter Elements Sling Set

PURPOSE: To lift, install and remove payload structural adapters in the horizontal and/or vertical attitudes.

DESCRIPTION: The sling set will consist of a variety of spreader bars, hooks, clevises, drop cables and straps which will be used to lift items for which specific sling sets have not been designated. This would include such items as test or service sets, shipping containers, and spacecraft for which spacecraft containers have not provided a special sling.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Weight
Fluid Media - N/A
Pressure - N/A
Capacity
Flow Rate - N/A
Size

INTERFACES: Payload adapters; building hoists; payload adapter element handling fixture.

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: None

REMARKS:

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE - Ground
ID NO. - H-H31
REQMT. - H-R31
ORIG. - H72-0736

NAME: Spacecraft Vertical Sling Set

PURPOSE: To provide, in conjunction with facility hoists, the capability to lift automated spacecraft and spacecraft combinations in the vertical attitude for assembly and installation on stands or onto the Tug/IUS.

DESCRIPTION: This set will consist of a variety of spreader bars, hooks, clevises, drop cables and straps which will be used to lift items for which specific sling sets have not been designated. This would include such items as test or service sets, shipping containers, and spacecraft for which spacecraft contractors have not provided a special sling. It will be capable of lifting loads up to 9072 kg (20,000 lbs) with diameters from 2 to 4.6 meters and lengths to 2 meters.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Weight - ____
Fluid Media - N/A
Pressure - N/A
Capacity
Flow Rate - N/A
Size

INTERFACES: Facility hoists; spacecraft hoist points; spacecraft rotation fixtures.

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: None

REMARKS:

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE - Ground
ID NO. - H-H33
REQMT. - H-R33
ORIG. - A70-0808

NAME: Work, Kickstage Assembly/Test Stand

PURPOSE: To support one or two STACKED KICKSTAGES and an automated spacecraft during integration, test and checkout, and servicing of the kickstage and kickstage/spacecraft assembly in the vertical attitude. The stand should have interchangeable provisions to mount, secure and restrain the kickstage while mounted on its kickstage to Orbiter, kickstage to kickstage, or kickstage to Tug/IUS flight adapters, or to secure the kickstage assembly directly via its main frame.

DESCRIPTION: The stand is a rigid frame which will support the tug and experiment during integration and checkout and will have interchangeable provisions to mount secure and restrain the kickstage while mounted on its adapters or to secure the assembly directly by its main frame.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Weight - TBD
Fluid Media - N/A
Pressure - N/A
Capacity - TBD
Flow Rate - N/A
Size - TBD

INTERFACES: Payloads, facilities

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: None

REMARKS:

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE - Ground
ID NO. - H-H34
REQMT. - H-R34
ORIG. - A70-0562

NAME: General Purpose Access Ladder

PURPOSE: To provide the capability to gain access to the payload component and/or element-handling fixture/sling interface locations and also to the environmental cover-sling interface location.

DESCRIPTION: The stand has a 0.6 m (2 ft) x 0.6 m (2 ft) adjustable work platform connected to a stairway which is positioned by a manually operated hydraulic pump. The stand will have wheels at one end of the base and casters at the other end for mobility. The stairway and platform will have hand rails.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Weight - TBD
Fluid Media - N/A
Pressure - N/A
Capacity - TBD
Flow Rate - N/A
Size - 4 x 1.5 x 4.6 m (13 x 5 x 15 ft)

INTERFACES: Aft bulkhead; rack/floor assembly; pallet, module

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: None

REMARKS: The following end items (with noted modifications) may also be used.

A72-1013 - Provide appropriate safety rails, be man-portable, and have wheel locking capability.

A70-0514 - Provide appropriate safety rails, and be man-portable

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE - Ground
ID NO. - H-H36
REQMT. - H-R36
ORIG. - H70-0802

NAME: Pallet Restraint Fixture

PURPOSE: To provide structural rigidity between independent pallets to allow handling/lifting them as a unit. The fixtures shall be attachable and/or removable when pallets are in the workstand or in the Orbiter bay. They shall allow attachment of independent pallets which are separated by TBD feet. They shall be designed to be rapidly removable such that their usage will have minimum or no impact on the overall Orbiter or STS time line. If necessary, they shall be equipped with appropriate lifting eyes to allow handling by sling devices.

DESCRIPTION: The fixture (strong-back) will be a rigid frame device consisting of beams, cables, attach hook devices and rings adjustable to accommodate varying lengths and shifting c.g.'s of payloads up to 4.6 m (15 ft) dia. by 18.3 m (60 ft) long and 29483 kg (65000 lb) weight. It will interface with the payload on a non-interference basis such that engagement and load transference to attachment/retention points can occur while the handling fixture is still attached. It will support an IUS/Tug with payload by attachment to the carrier only, and automated by attachment to the spacecraft or to a spacecraft-orbiter adapter. It will not induce any bending or twisting loads on any payload element.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Weight - 15876 kg (35000 lbs)
Fluid Media - N/A
Pressure - N/A
Capacity - 29483 kg (65000 lbs)
Flow Rate - N/A
Size - 18.3 x 5.2 x 2.1 m (60 x 17 x 7 ft)

INTERFACES: Payloads, payload cannisters,pads, P/L Room Spacelab pallets; sling assemblies.

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: None

REMARKS:

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE - Ground
ID NO. - H-H38
REQMT. - H-R38
ORIG. - A70-0834

NAME: Payload Work/Assembly Stand

PURPOSE: Provide a support structure for the Spacelab payload and/or payload elements during the performance of those activities associated with making up or dismantling a payload such as installation/removal of the rack/floor assembly into the experiment pressure shell, installation/removal of the crew access tunnel, and mating/demating of pallets to modules; the performance of any servicing and maintenance activities required on the payload or on the support module and its component parts; and the conduct of the Orbiter-Spacelab Interface Simulation Verification activities.

DESCRIPTION: The workstand is a horizontal support structure which supports and provides access to individual and integrated automated payloads. Retention fittings are similar in type, quantity and location to those provided in the Orbiter. Access is provided for the entire length of the workstand on both sides as well as the full width at both ends, for payload heights to 4.6 m (15 ft) and to all required positions within the envelope of the workstand. The stand includes cable trays for routing of electrical and fluid lines and interface panels for simulating the Orbiter-to-payload interfaces. Test equipment needed to perform Level 1 interface simulation is included.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Weight - TBD
Fluid Media - N/A
Pressure - N/A
Capacity - TBD
Flow Rate - N/A
Size - 3.0 x 5.5 x 1.8 m (10 x 18 x 6 ft)

INTERFACES: Spacelab, pallets, modules, tunnel, forward utility bridge, LPS, servicing units, installation/extraction fixtures.

MODIFICATION TO ORIGINAL PIECE OF EQUIPMENT: Provide interface equipment with LPS and appropriate service equipment, cable trays, fluid lines and an interface panel to simulate payload-to-Orbiter interconnects.

REMARKS:

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE - Ground
ID NO. - H-H39
REQMT. - H-R39
ORIG. - H72-0736

NAME: Spacelab Component Sling Set

PURPOSE: To be used, with a crane, to permit the raising and repositioning of the spacelab payload components (tunnel and aft bulkhead) from one position to another and of emplacing and removing a 15.2 m (50 ft) environmental cover from any spacelab payload element/component.

DESCRIPTION: The sling set will consist of a variety of spreader bars, hooks, clevises, drop cables and straps which will be used to lift items for which specific sling sets have not been designated. This will include such items as test or service sets, shipping containers, and spacecraft for which spacecraft contractors have not provided a special sling. It will accommodate loads varying in length from 2-4 meters, 1-5 meters wide, 1-5 meters high, and up to (TBD) Kg.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Weight - TBD
Fluid Media - N/A
Pressure - N/A
Capacity - TBD
Flow Rate - N/A
Size - TBD

INTERFACES: Spacelab tunnel, aft bulkhead, covers, facility crane.

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: None

REMARKS:

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE - Ground
ID NO. - P-H1
REQMT. - P-R1
ORIG. - P78-3103

NAME: Transportation Instrumentation Set

PURPOSE: To monitor and record payload environment data during transit at the launch site.

DESCRIPTION: The kit will provide instrumentation to monitor and record temperature, humidity, shock and vibration during the land and water transportation at the launch sites. The production instrumentation package (PIP) will be attached to the side of the transporters.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Weight - 45.4 kg (100 lbs)
Fluid Media - N/A
Pressure - N/A
Capability - N/A
Flow Rate - N/A
Size - 55.9 x 38.1 x 53.3 cm (22 x 15 x 21 i
Mobility - Portable

INTERFACES: Transporter, all automated payloads with or without IUS/TUG.

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: None

REMARKS:

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE - Ground
ID NO. - P-H2
REQMT. - P-R2
ORIG. - P70-0559

NAME: Payload Container Horizontal Transporter

PURPOSE: To transport the payload container in the horizontal attitude from payload processing facilities to the OPF or pad, and return.

DESCRIPTION: The transporter will support the loaded or unloaded payload container in the horizontal attitude. The unit shall have a flat bed [approximately 5.5 m (18 ft) wide by 19.8 m (65 ft) long] with tie-down provisions included. It will be towable from either end or by a prime mover, have steerable front and rear wheels, have self-contained braking and stabilization jacking provisions, and a suspension system to minimize over-the-road shock and vibration.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Weight - TBD
Fluid Media - N/A
Pressure - N/A
Capacity - TBD

Flow Rate - N/A
Size - 1.5 x 6.1 x 19.8 m (5 x 20 x 65 ft)
Mobility - Must be towable from either end

INTERFACES: Payload containers.

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: Provide vibration protection and capability to be towable from either end.

REMARKS:

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE - Ground
ID NO. - P-H3
REQMT. - P-R3
ORIG. - P70-0571

NAME: Payload Container Vertical Element Transporter

PURPOSE: To transport the payload container from the TPF or VAB to the pad and return, and to transport the payload element container between processing facilities.

DESCRIPTION: The transporter will support the loaded or unloaded payload container in the verticle attitude. It will also support the payload element container in the horizontal attitude. The unit will have a flat bed [approximately 5.5 m (18 ft) wide and 9.1 m (30 ft) long] with tie-down provisions enclosed. It will be towable from either end by a prime mover, have steerable front and rear wheels, have self contained braking and stabilization jacking provisions, and a suspension system to minimize over-the-road shock and vibration.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Weight - TBD
Fluid Media - N/A
Pressure - N/A
Capacity - TBD
Flow Rate - N/A
Size - 2.1 x 12.2 x 12.2 m (7 x 40 x 40 ft)
Mobility - Must be towable from either end

INTERFACES: Payload containers.

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: Provide front and rear towing and steering capability for braking and stabilization jacking; a suspension system to minimize over-the-road shock and vibration.

REMARKS:

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE - Ground
ID NO. - P-H5
REQMT. - P-R5
ORIG. - 470-0502

NAME: Universal Cover Set

PURPOSE: To cover payload element fluid and electrical connectors, and optical and other delicate component surfaces.

DESCRIPTION: The cover set provides the means of protecting the various fluid media openings from uncontrolled environments during transport, ferrying, storage, maintenance, and installation. Each cover set has provisions for moisture indication and if the system shall be required to breathe, a proper dessicant will be installed. Systems that can inadvertently be pressurized will contain relief mechanisms to preclude plug or closure expulsion. The covers are in general made from flexible and rigid plastic and metal. Each cover will not contribute to the contamination of the system being protected.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Weight - TBD
Fluid Media - N/A
Pressure - N/A
Capacity - N/A

Flow Rate - N/A
Size - TBD

INTERFACES: Payload elements

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: Modification to include covers for optics and other delicate component surfaces.

REMARKS:

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE - Ground
ID NO. - P-H11
REQMT. - P-R11
ORIG. - P70-1000

NAME: Component Handling Truck

PURPOSE: To move small items such as electronic racks and test equipment intrafacility at the launch site.

DESCRIPTION: The transporter is a standard 680 kg (1500 lbs) pick-up truck modified for towing or a factory type "mule" which is either battery driven or gasoline powered.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Weight - 1814.4 kg (4000 lbs)
Fluid Media - N/A
Pressure - N/A
Capacity - 680 kg (1500 lbs)
Flow Rate - N/A
Size - 1.5 x 1.5 x 3.7 m (5 x 5 x 12 ft)

INTERFACES:

MODIFICATION TO ORIGINAL PIECE OF EQUIPMENT: None

REMARKS:

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE: - Ground
ID NO. - P-H15
REQMT. - P-R15
ORIG. - P77-0018

NAME: Payload Component/Adapter Dolly

PURPOSE: To provide, in conjunction with a prime mover, a transport capability for selected payload components such as the Spacelab tunnel and various spacecraft structural adapters (spacecraft to Orbiter, spacecraft to kickstage, TUG/IUS to Orbiter, etc.) for movement within or between processing facilities.

DESCRIPTION: This dolly will protect, support and restrain its load during all transfer operations and shall interface either directly with the item or via an adaptive support stand or pallet. It will be capable of supporting up to 4536 Kg (5 ton), have a surface area of approximately 4.6 x 4.6 m (15 x 15 ft), contain attachment points to support use of a tie-down kit, have braking or wheel locking capability and be compatible with commercially available prime movers. This device will be used to transport its assigned load to and from storage as well as to transport returned adapters from the OPF to the refurbishment facility when these items are the only items returned from a mission.

It consists of a pneumatic-wheeled structure with attachment devices for securing a load for transporting or storage. The dolly will be tongue-steered and have parking brakes. It will have the capability of traversing nominal surfaced roads without damage to the loads.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Weight - 29030 kg (64,000 lbs)
Fluid Media - N/A
Pressure - N/A
Capacity - 54431 Kg (120,000 lbs)
Flow Rate - N/A
Size - 5.8 x 4.3 m (19 x 14 ft)
Height - TBD

INTERFACES: Prime mover; support stands and pallets; payload components and adapters.

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT:

REMARKS: The P77-0018, selected to satisfy the noted requirements has a capacity of 54431 kg (120,000 lbs) while requirement is for 4536 kg (10,000 lbs).

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE - Ground
ID NO. - S-H1
REQMT. - S-R1
ORIG. - S70-0613

NAME: Hydrazine Service Set

PURPOSE: To drain, flush, purge and fill as required the hydrazine systems of payloads at the TUG and hazard processing facilities.

DESCRIPTION: The set is a self-contained unit that contains fluid storage and refill capability, all plumbing and fittings, service hoses, gaging, pumps, regulators, valves, filters, and metering to accomplish fill, drain, flush and purge of payload hydrazine systems. Three separate systems are included so that MMF, N_2H_4 and N_2O_4 can be handled independently. The set is movable and includes provisions to utilize facility power and GN_2.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Weight - TBD
Fluid Media - MMH, N_2H_4, N_2O_4, (including provisions to utilize facility GN_2)
Pressure - TBD
Capacity - TBD
Flow Rate - TBD
Size - TBD
Elect. Power -

INTERFACES: Vehicle, payloads, facility power

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: Provide capability for three systems (MMH, N_2H_4, and N_2O_4); also to utilize facility power and GN_2.

REMARKS:

MMSE ITEM DESCRIPTION
SHUTTLE DERIVED

TYPE - Ground
ID NO. - S-H2
REQMT. - S-R2
ORIG. - C70-0743

NAME: Instrument Gas Service Set

PURPOSE: To supply instrument gas, as required, to orbiter payloads at the pad and at the processing facilities.

DESCRIPTION: The set is a self-contained unit that houses all the necessary tanks, valves, regulators, filters, flex lines and fittings to accomplish instrument gas transfer to required payloads. Tanks will be sized to allow full servicing with at least 50% reserve. The system is equipped with variable flow and pressure capacity, automatic and manual safety relief valves, a system status display panel, a gas filtering system and tank refill capability. The set is portable.

PHYSICAL/FUNCTIONAL CHARACTERISTICS:

Weight -
Fluid Media - GN_2 or GH_e
Capacity - TBD
Flow Rate - TBD
Size - TBD
Elect. Power - Utilize facility power
Pressure - Six pressure ranges are as follows:

Flow Rate - TBD
Size - TBD

Config.	Gage Pressure		Working Range	
	Pascals	Psig	Pascals	Psig
-001	0-2068	(0-30)	0-1657	(0-24)
-002	0-6895	(0-100)	1379-5516	(20-80)
-003	0-20684	(0-300)	4137-16547	(60-240)
-004	0-55156	(0-800)	11031-44125	(160-640)
-005	0-137890	(0-2000)	27578-11031	(400-1600)
-006	0-413670	(0-6000)	82734-330936	(1200-4800)

The -001 through -004 configurations have two stages of regulation whereas the -005 and -006 have only one stage of regulation.

INTERFACES: Vehicle, payloads, facility power.

MODIFICATIONS TO ORIGINAL PIECE OF EQUIPMENT: Provide instrument servicing and make portable.

REMARKS:

AIRBORNE REQUIREMENTS/EQUIPMENT

REQUIREMENT			EQUIPMENT			
SD ID	MSFC MMSE ID	Title	SD ID	Equipment Origin	Applicability	Disposition Rationale; Comments
		1.3.3 POWER				
		<u>1.3.3.1 Electrical Power Generation (EP)</u>				
EP-R1		Auxiliary Power Unit (Tug and IUS)	EP-H1	IUS battery	As is	Additional batteries can be added.
EP-R2		Auxiliary Power System	EP-H2		New	Large portion of system components assumed to come from Shuttle program.
		1.3.4 AVIONICS				
		<u>1.3.4.1 Guidance, Navigation & Control (GN&C)</u>				
GN-R1		Small IPS (miniaturized pointing mount)	GN-H1	SIPS	Mod.	Goddard SIPS may be candidate.
GN-R4		Celestial Sensor-Coarse (10-30 arc-sec)	GN-H4	MC-431-0128	Mod.	Provide 95-mm lens, lightshade and bright object detector.
GN-R5		Celestial Sensor-Fine (1.0 arc-sec)	GN-H5	MC-431-0128	Mod.	Provide a 456-mm lens, lightshade and bright object detector.
GN-R6		Earth (Horizontal) Sensor (180-360 arc-sec)	GN-H6	MC-432-0214 (GPS)	As is	Global Positioning Satellite hardware.
GN-R9		Payload Integrated Pointing System	GN-H9		New	Portion of system components assumed to come from Shuttle program.
		<u>1.3.4.2 Communications & Tracking (CT)</u>				
CT-R2		TV Camera (commercial)	CT-H2	ICD-3-0050-01	As is	
		<u>1.3.4.3 Displays & Controls (D&C)</u>				
DC-R1		Payload Specialist Station (PSS)	DC-H1		New	Portion of system components assumed to come from Shuttle program.
		<u>1.3.4.4 Instrumentation (IN)</u>				
IN-R1		Protective Device - Earth/Moon/Sun Sensor	IN-H1	MC-431-0128	As is	The protective device used with the Orbiter star tracker can be used.
DP-R1		<u>1.3.4.5 Data Processing & Software (DP)</u>				
		Mini/Micro Computer	DP-H1	Autonetics MICRON 16	As is	The computer being investigated at Autonetics looks good
DP-R2		Payload Multiplexer/Demultiplexer	DP-H2	MC-615-0004	Mod.	MDM will be redesigned as a modulator-type unit.
		1.3.5 ENVIRONMENTAL CONTROL, LIFE SUPPORT				
		<u>1.3.5.3 Thermal Control (TC)</u>				
TC-R1		RTG Cooling Unit	TC-H1		New	Portion of system components assumed to come from Shuttle program.

AIRBORNE REQUIREMENTS/EQUIPMENT (continued)

REQUIREMENT			EQUIPMENT			
SD ID	MSFC MMSE ID	Title	SD ID	Equipment Origin	Applicability	Disposition Rationale; Comments
		1.3.6 CREW STATION & EQUIPMENT				
		<u>1.3.6.2 Crew Provisions & Accommodations (CA)</u>				
CA-R1		Payload Work Stations	CA-H1		As is	Basic Orbiter plus planned mission kits include necessary work stations.
		1.3.7 MECHANICAL SYSTEMS				
		<u>1.3.7.7 Payload Bay Systems (PB)</u>				
PB-R4	SP-1	Special Pallet	PB-H4	Spacelab	Mod.	Spacelab pallet increased capability to 19,000 lb.
PB-R5	SP-2	Special Pallet	PB-H5	Spacelab	Mod.	Spacelab pallet increased capability to 16,500 lb.
		NO WBS CATEGORY				
NC-R1		Purge System	NC-H1	A34-0364	None	Mods too extensive for flight use.
NC-R23	PDK-1	Payload Docking Kit-Probe	NC-H23	Apollo	Mod.	Manual mechanisms should be removed.
NC-R24	PDK-2	Payload Docking Kit-Drogue	NC-H24	Apollo	As is	
NC-R38	PUS-1	Payload Umbilical Support	NC-H38	V70-340633	As is	

GROUND REQUIREMENTS/EQUIPMENT

REQUIREMENT			EQUIPMENT			
SD ID	MSFC MMSE ID	Title	SD ID	Equipment Options	Applicability	Disposition Rationale; Comments
		1.9.1 GROUND SUPPORT EQUIPMENT AUXILIARY (A)				
A-R1	KMA-MH-44	Environmental Conditioning Unit	A-H1	S70-0573*	Mod.	Two units req'd; add power generation system
				S78-0108	Mod.	Add power generation system
				S70-0707	Mod.	Add power generation system
				S70-0708	Mod.	Add power generation system
A-R2	KMB-MS-09	Payload Purge Cart	A-H2	A34-0364*	Mod.	Add gas bottles and trailer tongue
A-R11	SH-18	Personnel Air Conditioning Unit	A-H11	S70-0707	As is	
				S70-0708	As is	
				S78-0108	As is	
				S70-0573*	As is	Two units may be required
A-R12	AH-38	P/L Bay Mobile Air Conditioning Unit	A-H12	See A-R1		
A-R13	TS-12	RTG Cooling Unit	A-H13	New	As is	
A-R14	ST-04	Ground Power Supply Set	A-H14	C14-0262*	Mod.	Provide 4-kW capacity
				C70-0693*	As is	
		Checkout (C)				
C-R2	KMC-MS-11	Set, Alignment, Spacecraft G&N	C-H2	C70-0701*	Mod.	Provide rough orientation, specific sun sensor capability, and gyro and momentum wheel functional verification.
C-R3	KMC-MS-12	Set, Calibration, Spacecraft Electronics	C-H3	C70-0716*	None	No calibration capability
				C70-0721*	None	No provisions for "S" and "X" bands
				C70-0727*	None	No calibration capability
C-R4	KMC-MT-08	Cable Set	C-H4	C70-0519	As is	
				C72-1049	As is	
				C72-1055*	As is	
C-R5	KMC-MT-09	Set, Breakout Boxes	C-H5	C70-1087*	As is	
C-R6	KMC-MT-10	Ordnance Circuits Simulator	C-H6	H72-0736	None	Does not determine whether adequate ordnance device firing signals exist
C-R7	KMC-MT-11	Simulator, Payload Electrical	C-H7	C70-0547	As is	
C-R8	KMC-MT-12	Test Set, Communications and Instrumentation	C-H8	C70-0565*	As is	
				C70-0584*	None	No system-level checkout capability
				C70-0646*	None	No communications checkout systems capability
C-R9	KMC-MT-13	Test Set, Spacecraft Engine Alignment	C-H9	A70-0645*	Mod.	Provide for attitude control thrusters alignment and calibration
C-R10	KMC-MT-14	Spacecraft G&N System Test Set	C-H10	C70-0701*	None	No capability for functional operational tests
C-R11	KMC-MT-15	Test Set, Spacecraft Power System	C-H11	C70-0656*	None	For electrically actuated valves only
				C70-0657*	None	Partial check of specified equipment

*Space Division design responsibility.

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Space Division
Rockwell International

GROUND REQUIREMENTS/EQUIPMENT (continued)

REQUIREMENT			EQUIPMENT			
SD ID	MSFC MMSE ID	Title	SD ID	Equipment Options	Applicability	Disposition Rationale; Comments
			C-H11	C70-1050*	Mod.	Provide adequate electrical interfaces and
			(cont.)	C70-0547	Mod.	required electrical connectors
C-R12	KMC-MT-16	Test Set, Spacecraft Propulsion System	C-H12	C70-0647*	Mod.	Provide spacecraft propulsion system checkout
C-R17	AS-07	Memory Load and Verify Unit	C-H17	C70-0701*	None	
C-R18	TS-10	Spacecraft Memory Load and Verify Unit	C-H18	C70-0701*	None	
C-R19	MT-06	PSS Simulator	C-H19	A70-0834	None	No functional capability
				A70-0810	None	No functional capability
				A70-0811	None	No functional capability
C-R22	AT-10	Payload Antenna Test Hat Set	C-H22	C70-0565*	As is	
				C70-0584*	None	No communications system checkout
				C70-0646	None	capability
C-R25	TT-04	Spacecraft Antenna Dummy Load Set	C-H25	C70-0565-1	As is	
				C70-0565-2	As is	
		HANDLING (H)				
H-R1	KMA-MH-03	P/L Container Horizontal Access Equipment	H-H1	A70-0519*	None	Used to work on/in cargo bay; not P/L
				A70-0835	None	Deleted from GSE inventory
H-R3	KMA-MH-10	P/L Container	H-H3	A70-0806	As is	
H-R4	KMA-MH-11	P/L Element Container	H-H4	A70-0806	Mod.	Redesign smaller container
H-R5	KMA-MH-19	P/L Handling Fixture	H-H5	A70-0802	As is	
H-R6	KMA-MH-27	Multipurpose Sling Set	H-H6		As is	Set can be made from 57 sling sets in Shuttle program
H-R7	KMA-MH-29	P/L Container Sling Set	H-H7	H70-0804	As is	
				H78-3007	Mod.	Add attach points (ET design)
				H78-3006	Mod.	Add attach points (ET design)
				H78-3004	Mod.	Add attach points (ET design)
H-R8	KMA-MH-34	S/C Vertical Assembly Stand	H-H8	A70-0808	As is	
H-R10	KMA-TH-55	Tug/Payload Handling Fixture	H-H10	H70-0802	As is	
H-R11	KMA-MT-07	Payload Mechanical Simulator	H-H11	A70-0807	As is	
H-R12	KMB-AH-30	P/L Assembly/Test Horiz. Work Stand	H-H12	A70-0834	No	No interface panels for simul.
				A70-0810	Mod.	Provide universal functional and physical
				A70-0811	Mod.	interface
H-R15	MH-14	Mobile Crane	H-H15	H70-0692	As is	
				U72-1138	As is	
H-R19	MH-05	P/L Assy/Test Work Stand	H-H19	A70-0810	Mod.	Provide means of access to the interior of the
				A70-0811	Mod.	workstand
H-R20	MH-31	Horizontal S/C Sling Set	H-H20	See H-R6	--	
H-R21	MH-32	S/C Shipping Container Sling Set	H-H21	See H-R6	--	
H-R22	SH-04	Workstand Access Platform	H-H22	See H-R19	--	
H-R23	AH-16	P/L Assy/Test Stand Access Platform	H-H23	See H-R19	--	
H-R24	AH-29	P/L Strongback Sling Set	H-H24	See H-R7	--	
H-R27	TH-40	P/L Sling Set	H-H27	See H-R6	--	

* Space Division design responsibility

GROUND REQUIREMENTS/EQUIPMENT (continued)

REQUIREMENT			EQUIPMENT			
SD ID	MSFC MMSE ID	Title	SD ID	Equipment Options	Applicability	Disposition Rationale; Comments
H-R28	MH-02	P/L Bay Horizontal Access Equipment	H-H28	A70-0519	Mod.	Provide capability to support 2 men on all access structure
H-R30	MH-28	P/L Adapter Elements Sling Set	H-H30	H72-0736*	As is	
H-R31	MH-30	Spacecraft Vertical Sling Set	H-H31	H72-0736*	As is	
H-R33	MH-35	Kickstand Assy/Test Workstand	H-H33	A70-0808	As is	
H-R34	SH-01	General Purpose Access Ladder	H-H34	A72-1013	Mod.	Provide for man portability
				A70-0562	As is	
				A70-0514	Mod.	Provide safety rails and for man-portability
H-R36	SH-07	Pallet Restraint Fixture	H-H36	H70-0802	As is	
H-R38	SH-13	P/L Work Assembly Stand	H-H38	A70-0834	Mod.	Include interface equipment to LPS, service equipment and cable trays
H-R39	SH-16	S/C Components Sling Set	H-H39	H72-0736*	As is	
		PACKAGING & TRANSPORT (P)				
P-R1	KMA-MH-26	Transportation Instrumentation Set	P-H1	P78-3103	As is	
P-R2	KMA-MH-39	P/L Container Horizontal Transporter	P-H2	P70-0559	Mod.	Provide capability to tow from either end
P-R3	KMA-MH-41	P/L Container Vertical/Element Transporter	P-H3	P77-0006	None	Does not provide for horizontal transport
				P70-0571*	Mod.	Provide front and rear towing and steering
P-R5	KMC-MH-46	Universal Cover Set	P-H5	A70-0502*	Mod.	Include covers for optics & other delicate component surfaces
P-R10	MH-24	Interior Prime Mover	P-H10	P70-0802	None	Insufficient pulling capacity
P-R11	MH-42	Component Handling Trck	P-H11	P70-1000	As is	
P-R13	MH-40	Vertical P/L Container Transporter	P-H13	See P-R3		
P-R15	MH-16	P/L Component/Adapter Dolly	P-H15	P77-0018	As is	
		SERVICING (S)				
S-R1	KMB-MS-01	Hydrazine Service Set	S-H1	S70-0613*	Mod.	Provide three separate systems
S-R2	KMB-MS-02	Instrument Gas Service Set	S-H2	C70-0734*	Mod.	Provide instrument servicing capability and make portable

* Space Division Design Responsibility

APPENDIX A4

MULTI-DISCIPLINE AUXILIARY PAYLOAD POWER SYSTEM

INTRODUCTION

The Space Processing Application (SPA) discipline has for several years been showing power requirements greater than can be supplied by the Orbiter. Since this was a payload-unique requirement, the auxiliary power system was to be provided by the SPA payloads themselves. In the 1975 version of the SSPD several other payload disciplines have also indicated the need for power above the 4.0 - 5.2 kw net (plus 1.8 - 3.0 kw required for Spacelab systems) available to the Spacelab payloads or experiments. In addition, there are logical combinations of payloads (mixed) which will require power at levels exceeding those normally available from Orbiter. The conceptual design of the MAPPS concept arose from a company-sponsored effort originally intended to provide a cheaper, more integrated alternative to the MSFC Auxiliary Payload Power System (APPS) concept. The work has been utilized here to define the special emphasis MMSE item.

OBJECTIVE

The objectives of this special emphasis task are:

1. To define the concept for multi-discipline auxiliary payload power system (MAPPS) which can provide the additional power and energy required by the payloads.

2. To accomplish this with minimum impact on the payload bay volume at minimum cost and weight.

APPROACH

The study was limited to the use of current Orbiter systems to their maximum capacity, to batteries and to fuel cells as the power source. Fuel cells are the prime consideration because of the capability and their availability from the Orbiter program, thus eliminating development costs.

The concept for MAPPS was defined by:

1. Reviewing the payloads power and energy requirements to determine the range of capability required,

2. Investigating the various methods of obtaining the needed power and energy, and selecting the optimum for the application identified,

3. Identifying the heat rejection options compatible with the total power being used by the payload, comparing these options and selecting the optimum consistent with Orbiter payload bay contamination and payload viewing constraints.

4. Determining the mass properties.

RESULTS

Task 1 - Requirements Identification

The 1975 SSPD lists two payloads in addition to the SPA payloads having power requirements which can be satisfied only with the auxiliary power kit. Combined payloads, while not defined at this time, must definitely be given serious consideration as evidenced by many studies and references such as the following excerpt from the final report of the Space Shuttle Payload Planning Working Group Volume 9 (5-73).

> ". . . since most of the development and operating costs of the Shuttle will be associated with its ability to lift weight into orbit, consideration should be given to organizing onboard utilities such as power, data systems, radiators, etc., so that full lifting capacity can always be utilized."

The "Manned Orbital Systems Concept" (MOSC) study (Reference 9)* has identified 18 combined payloads. Six of these combinations have average power ranging from 4-8 kw and 2 combinations require greater than 15 kw steady state with energy requirements up to 3960 KWH. Table 1 lists these 8 MOSC combined payloads and the high power user payloads from the SSPD with their respective power and energy requirements.

Task 2 - Power Generation Concept Definition

Two major power generation options were considered. Category 1 - Limited Capability, with three sub options, and Category 2 - Full Capability Fuel Cell Concepts, with three sub options (Figure 1).

Category 1 - Limited Capability

Currently, the Orbiter average power load while on orbit is 11-12 kw. Thus, 2-3 kw is potentially available from Orbiter fuel cells #1 and #2. Two problems arise when this option is considered. First, the Orbiter system power requirements may increase, thus eliminating this source of extra power and second, the Orbiter electrical system is not currently configured to permit fuel cells #1 or #2 to be attached to the payload bus nor is there any means to transfer power from the Orbiter systems to payloads. For these reasons this option was considered inadvisable and was dropped from further consideration.

*Appendix A1

Table 1. High Payload Power Requirements

Payload	Power (kw) Nominal	Power (kw) Peak	Energy (kwh)	No. of Missions
<u>Space Processing</u>				39
SP-01	5.7	9.3	230	
SP-02	6.8	18.1	362	
SP-03	7.8	21.4	303	
SP-04	3.5	4.9	175	
SP-14	13.6	22.0	1639	
SP-15	9.9	21.5	1188	
<u>Earth Observations</u>				
EO-20	5.0	7.5	138	12
<u>Atmospheric & Space Physics</u>				
AP-06	5.8	10.0	1275	29
<u>Combined Payloads</u>				
C-4	4.6			
C-6	5.1			
C-5	5.0			
C-7	4.9			
C-9	4.6	5.8	481	
C-13	16.4	50.1	3960	
C-12	15.4	42.8	3185	
C-17	8.0			

Volume XIV of JSC 07700 "Payload Accommodations" states that fuel cell #3 (payload-dedicated) can provide 7.0 kw nominal and 12.0 kw peaks for 15 minutes every three hours. The fuel cells will be qualified to a power profile which includes a 12 kw peak for 1 hour and an 8 kw peak for 5 hours, every 30 hours. While these longer power peaks are potentially available it was considered advisable to drop this concept from further consideration because these qual limits represent a growth margin for the Orbiter subsystems which should remain as such.

A modular battery kit concept was investigated. The final concept has the option of 1 to 4 batteries at 3 kw and 18 kwh maximum output for each battery. Figure 2 shows the method of modularizing the system and the weight for each of the four kits. At the present time an emergency battery kit is being designed for the Orbiter Flight Test Program (OFT) and the concept shown in Figure 2 is identical; thus, the capability is expected to be available for those payloads which require high power (less than 19 kw nominal) but which have low energy requirements, as the maximum energy level of the four battery configuration is only 72 kwh. The operating characteristics of the battery chosen (Apollo/Skylab) are:

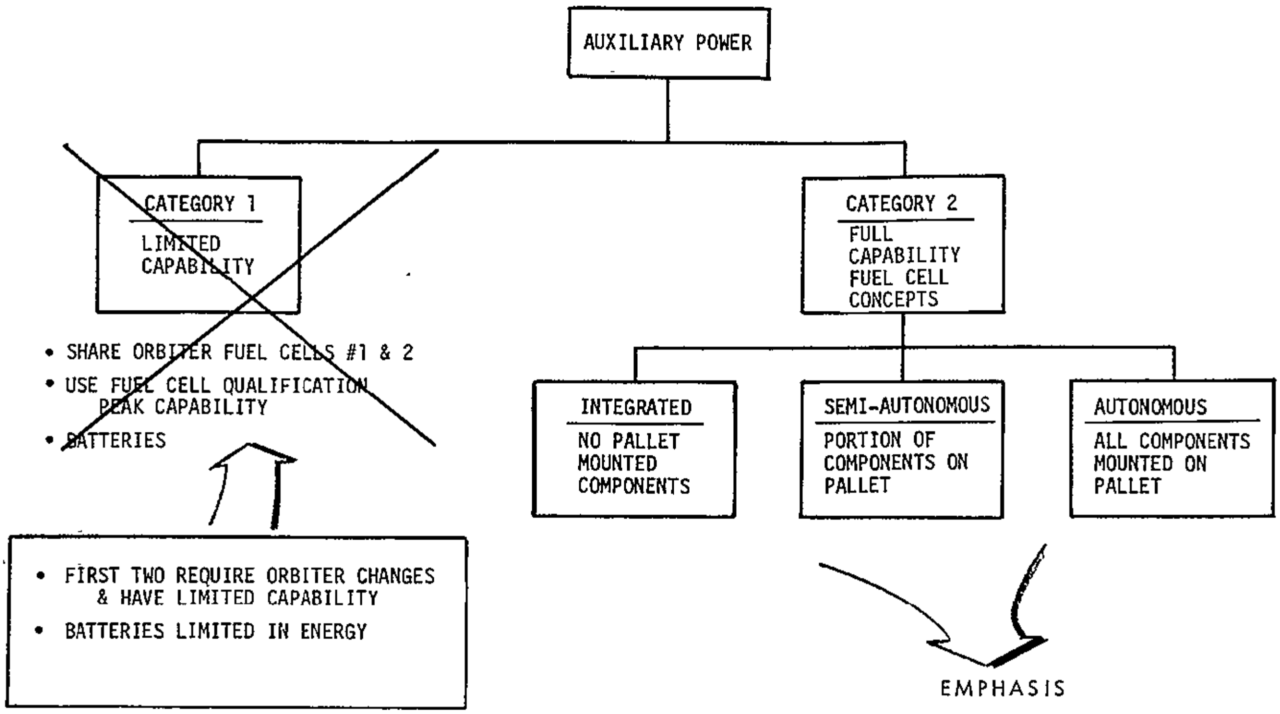


Figure 1. Power Generation Options

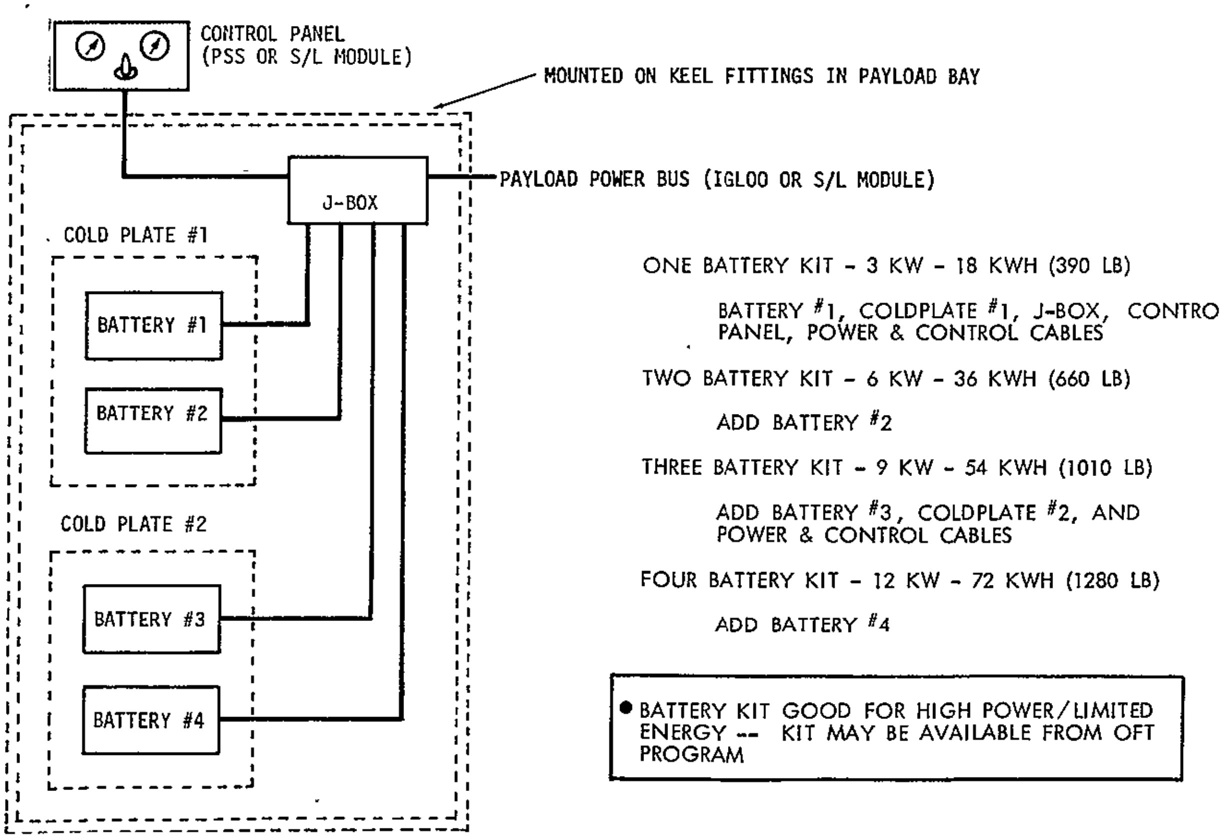


Figure 2. Modular Battery Kit Concept

- Ag Zn
- Reusable (secondary)
- 600 AH @ 35 A
- 123 kg (270 lbs)
- Non-operating temp 0 - 27°C (32 - 80°F)
- Operating temp - 93°C (200°F)
- Requires design change for cold plate mounting

Category 2 - Full Capability Fuel Cell Concepts

Since one of the objectives was to minimize cost and weight and maximize payload bay volume an investigation was made to determine the optimum location of each of the major power generation components (fuel cells, cryo tanks and water tanks). The autonomous concept is the extreme case, in regard to use of payload bay volume. Thus, the goal was to remove the components from the dedicated structure and move toward the semi-autonomous and integrated concepts. The best location for the fuel cells was to mount them on specially designed keel fittings which would be mounted in the two forward-most locations in the payload bay. These locations are ahead of a five pallet configuration of Spacelab and under the tunnel of a Spacelab using a module. The new fuel cells #4 and #5 will be plumbed into the cryo supply system at the same location as the three Orbiter fuel cells. The location of the MAPPS fuel cells (almost directly over the Orbiter cells) permits the cryo lines to be very short. The standard Shuttle extended mission cryo tanks and installations will be used, thus space has already been allocated (underneath the liner). The product water from the fuel cells will be integrated with the Orbiter fuel cell product water tanks. The selected location and/or use of Orbiter systems for the power generation components requires no usable payload bay volume. The results of this component location investigation are shown in Table 2 along with the advantages of mounting each component on a dedicated pallet and in the selected location. The table also lists the radiators and flash evaporator and are shown here for completeness as they are the two reaming major MAPPS components. Task 3 which follows, will provide the rationale for selection of the radiators and flash evaporator. As noted from Table 2 the selected location for the components results in a semi-autonomous system concept. Figure 3 shows the relative location within the Orbiter of the major power generation components.

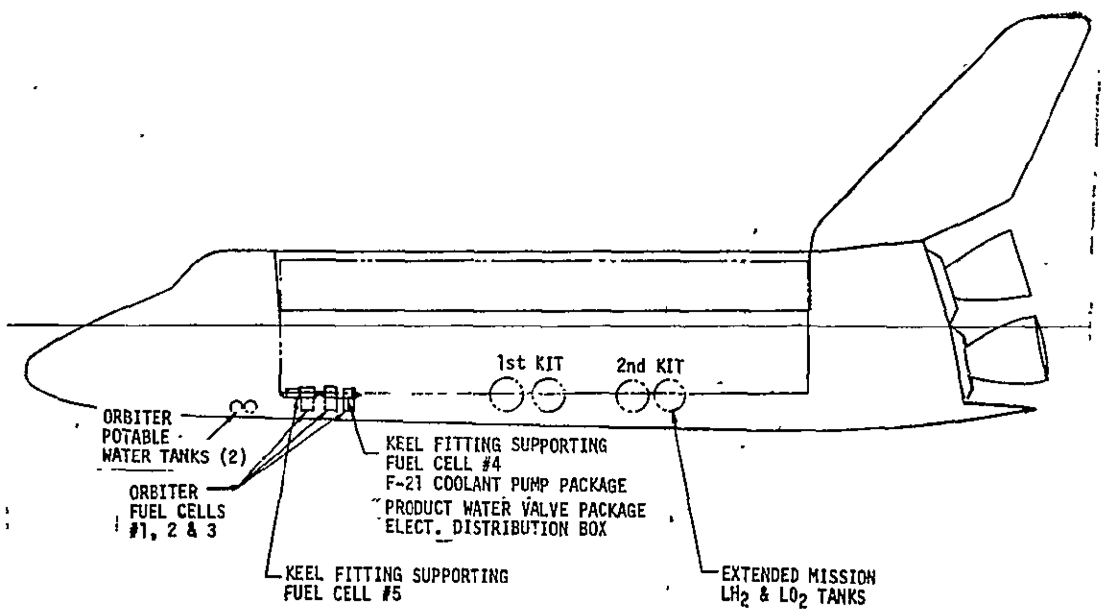
The Orbiter electrical system wiring to the payloads is being designed for 12 kw max (max output of one fuel cell). The Spacelab electrical system is also being designed for 12 kw max loads. However, with three fuel cells operating simultaneously at maximum capacity 36 kw will be generated which is considerably above the Orbiter and Spacelab capacity. To solve this problem the power distribution and control (PDC) box of the MAPPS kit will be utilized as a common interface point for all payload power. The power generated by Orbiter fuel cell #3 will be routed through the standard Orbiter system to the payload power interface at STA 695 and then by payload cables to the MAPPS PDC box. This box will assure proper sharing of fuel cell outputs and payload power loads. The power required by the Spacelab will be routed directly to the Spacelab with a separate cable for the experiments which requires the high power. This concept is graphically portrayed in Figure 4.

c.3

Table 2. Advantages of Major Component Locations

MAJOR COMPONENT	MOUNT ON DEDICATED PALLET	ALTERNATE LOCATION
RADIATORS	• NO INSTALLATION EFFORT DURING ORBITER TURNAROUND PERIOD (OFF LINE)	- STANDARD ORBITER BRIDGE FITTINGS • CAN BE MOUNTED TO PAYLOAD PALLET IF DESIRABLE • SMALL PAYLOAD VOLUME IMPACT (≈ 4%)
FLASH EVAPORATOR	• NO INSTALLATION EFFORT DURING ORBITER TURNAROUND PERIOD (OFF LINE)	- ON RADIATOR SUPPORT STRUCTURE • NO ORBITER SYSTEM IMPACT • MINOR PAYLOAD VOLUME IMPACT
FUEL CELLS	• NO INSTALLATION EFFORT DURING ORBITER TURNAROUND PERIOD (OFF LINE)	- KEEL FITTINGS IN FORWARD PORTION OF BAY (STA 636 TO 693) • CLOSE TO EXISTING FUEL CELL, CRYO TANK & OTHER INTERFACES • NO PAYLOAD VOLUME IMPACT
WATER TANKS	• INDEPENDENT FROM ORBITER SYSTEMS	- SHARE EXISTING ORBITER SYSTEM • NO ADDITIONAL HARDWARE • NO PAYLOAD BAY VOLUME
CRYO TANKS	• NO INSTALLATION EFFORT DURING ORBITER TURNAROUND PERIOD (OFF LINE)	- STANDARD CRYO KIT TANK LOCATIONS • NO ADDITIONAL HARDWARE • NO PAYLOAD BAY VOLUME • STANDARD SERVICING PROVISIONS

● SEMI-AUTONOMOUS CONCEPT WITH ONLY THE RADIATORS AND FLASH EVAPORATOR STOWED WITHIN BAY APPEARS BEST



- MAXIMUM VOLUME AVAILABLE FOR PAYLOADS/EXPERIMENTS
- SHORT, INTEGRATED CRYO CONNECTIONS
- UTILIZATION OF EXISTING VARIABLE CRYO KIT TANKAGE

Figure 3. Semi-Autonomous Concept Configuration

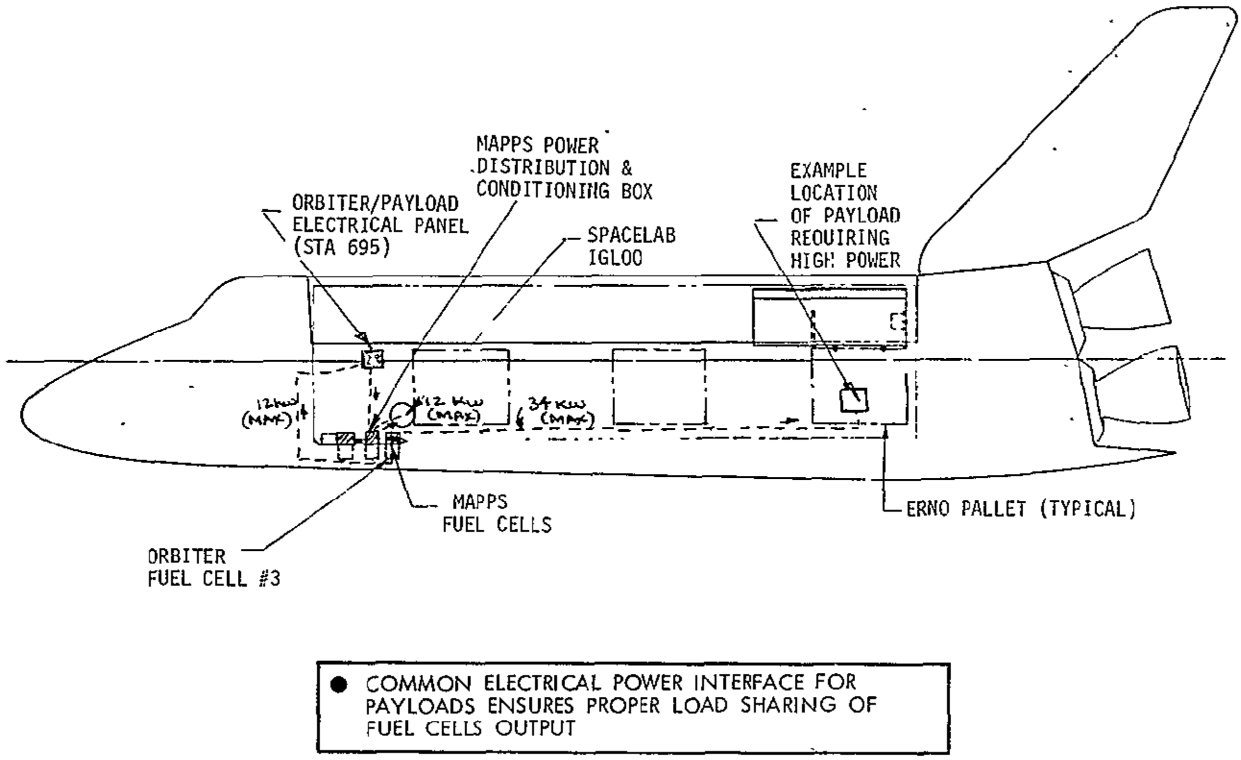


Figure 4. Power Kit Electrical Distribution Concept

Task 3 - Heat Rejection Concept Definition

The total heat rejection requirements come from two sources, heat directly from the fuel cell during their inefficient conversion of liquid O_2 and H_2 to electrical power and the heat generated by the use of the electrical power produced.

Two major options were considered, each with three sub options.

1. Use of Orbiter systems

   - Selective orientation
   - Orbiter flash evaporators
   - Increase coolant flow rate (pump kit)

2. Orbiter-independent

   - Fixed radiators
   - Deployable radiators
   - Deployable radiators with flash evaporator

The Orbiter Active Thermal Control System (ATCS) has the potential of rejecting additional head (above the design nominal) by making use of selective orientation of the vehicle. Both the vehicle roll angle and the orbit

inclination angle have a significant effect (roll ≈ 50% increase and inclination ≈ 30% increase) on the efficiency of the radiators. This concept has been eliminated from further consideration because it cannot satisfy the peak demands and because some payloads may have orientation requirements which would not permit the Orbiter to change its orientation.

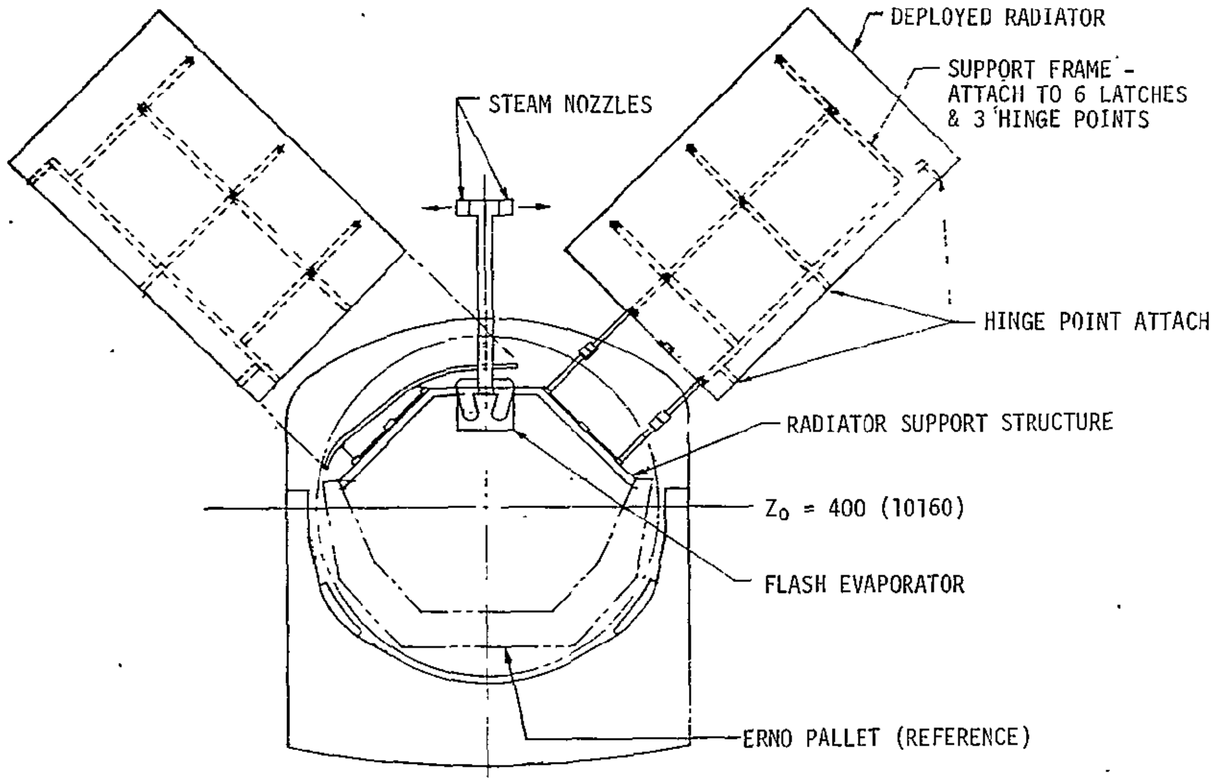
The Orbiter flash evaporator is composed of two separate sections, one for on-orbit use (7650 kg Cal/hr (39000 Btu/hr)), and a second section (25,200 kg Cal/hr (100,000 Btu/hr)) which in combination with the first is used during ascent and descent. The 25200 kg Cal/hr (100,000 Btu/hr) unit is vented overboard through a single nozzle (propulsive). This unit may be slightly undersized to handle the peak power (36 kw) heat rejection requirements of MAPPS but probably would be adequate. Its on-orbit use would require hardware changes to the Orbiter with the accompanying scar weight penalty. In addition, the continuous steam vent may be unacceptable to many contamination-sensitive payloads. Therefore, this concept will be given no further consideration.

A third concept which could provide additional heat rejection capability is to add a pump package to the ATCS Freon System. This would increase the flow rate in the system and thus increase the heat rejection capability by approximately 4540 kg Cal/hr (18000 Btu/hr). The second pump package would increase the system operating pressure necessitating an investigation of all system components to determine the need for requalification and possibly redesign. For this reason and also because it, too, does not have sufficient capacity, this concept has been dropped.

The Orbiter-independent systems, as the title implies, do not require changes to the basic Orbiter systems. The radiator panels to be used are duplicates of the forward radiator panels of the ATCS and are stowed either permanently or during ascent and descent in the case of the deployable type. Even in the deployable configuration (both sides rejecting heat) the two panels (9324 kg Cal/hr (37000 Btu/hr each)) do not have sufficient capacity during the 36 kw peak power generating periods, thus the selection of an Orbiter flash evaporator unit as the mans of satisfying the peak loads of approximately 26460 kg Cal/hr (105,000 Btu/hr). The steam vent is also deployed and consists of a non-propulsive 18-20 cm (7-8 in.) nozzle. In their stowed position the radiators require approximately 76 cm (30 in.) of the top portion of the aft 4.6 m (15 ft) of the payload bay. The deployed nozzle is aft of the radiator panels which are deployed vertical to the payload bay at approximately station 1295. Therefore, the steam exhaust is not expected to create a contamination problem for the payloads. Figures 5 and 6 show the installation of the radiators and flash evaporator. These figures indicate the almost negligible payload bay volume required by the system in its deployed configuration. The flash evaporator unit intrudes into the bay approximately 76 cm (30 in.). This creates practically no vertical viewing constraint for the payloads.

Task 4 - Mass Properties

A detailed system weight breakdown is shown in Table 3.



- PROVIDES DELTA HEAT REJECTION CAPABILITY OF 37,000 Btu/HR PER PANEL
  - SEPARATE FROM ORBITER SYSTEM
  - REQUIRES SMALL AMOUNT OF PAYLOAD BAY VOLUME DURING ASCENT & DESCENT

Figure 5. Deployable Radiator Kit Installation
(Looking Aft)

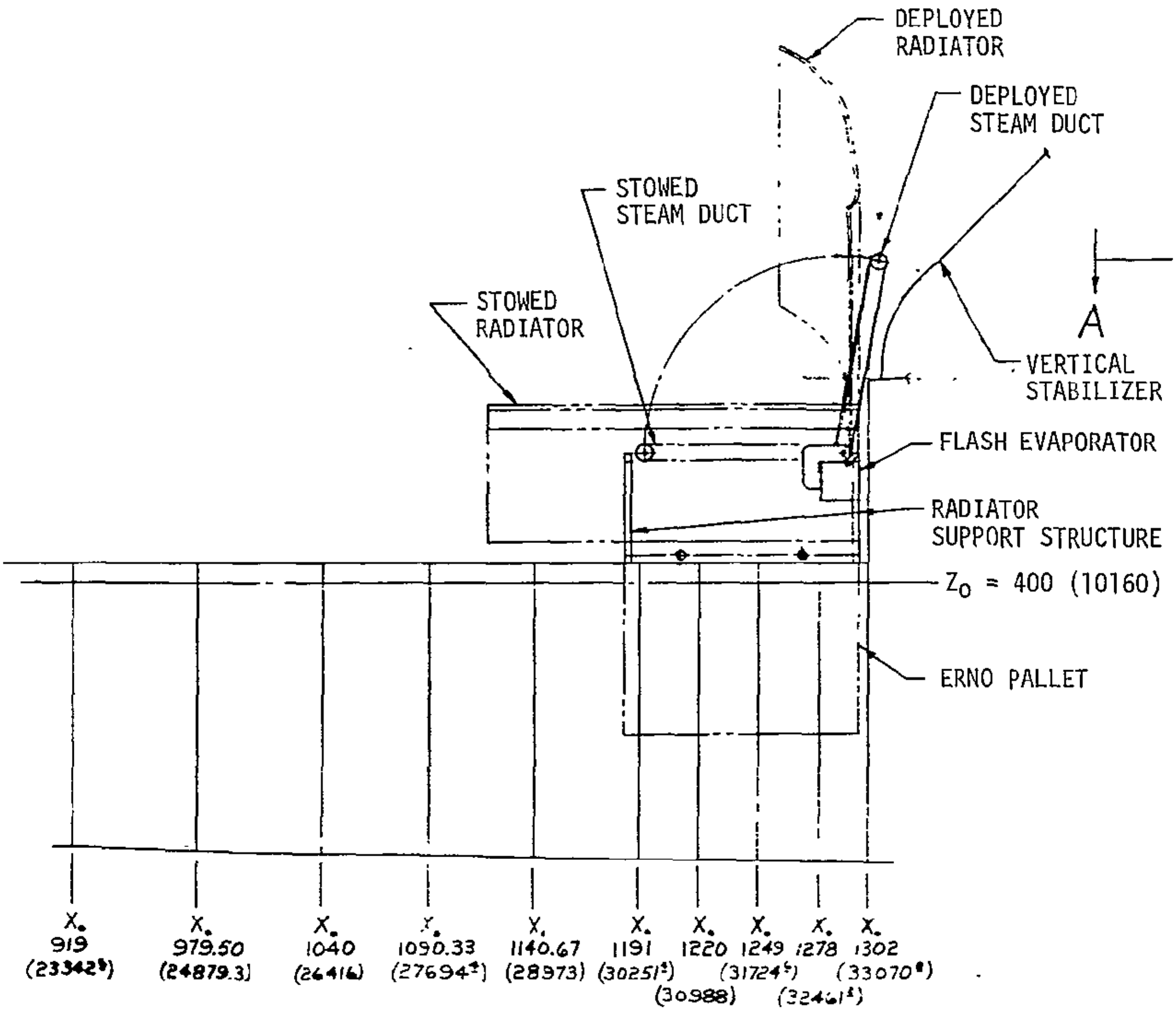


Figure 6. Deployable Radiator Kit Installation
(Side View)

Table 3. System Weight Breakdown

Component	Weight - Kg	(lbs)
Fuel cell (2)	185.3	(408)
Product H_2O value package (2)	2.7	(6)
Power conditioning system	113.8	(250)
Control panel	6.8	(15)
Power distribution box	20.4	(45)
Electrical cables	25.0	(55)
Fluid lines	9.1	(20)
Keel fittings (2)	43.1	(95)
Flash evaporator (1)	22.7	(50)
Flash evaporator installation	22.7	(50)
Cooland pump package (1)	50.5	(111)
Radiators (2)	177.1	(390)
Radiator support structure	118.1	(260)
Cryo storage (2 kits)	1169.5	(2572)
Total System Weight	1968.8	(4327)

CONCLUSIONS

There is a definite need for a multi-discipline auxiliary payload power system as evidenced by the 80 missions, identified in the SSPD, which require more power than can be supplied by the Orbiter. The as yet undefined combined payloads further justify the need for an MMSE kit to satisfy their power requirements.

The MAPPS configuration shown schematically in Figure 7 and pictorially in Figure 8 provide the needed capability at a very low development cost of $750,000 and low unit cost of $2,950,000. Its system weight is over 1500 kg (3300 pounds) lighter than the similar APPS and only requires at most 76 cm (30 in.) of payload bay volume.

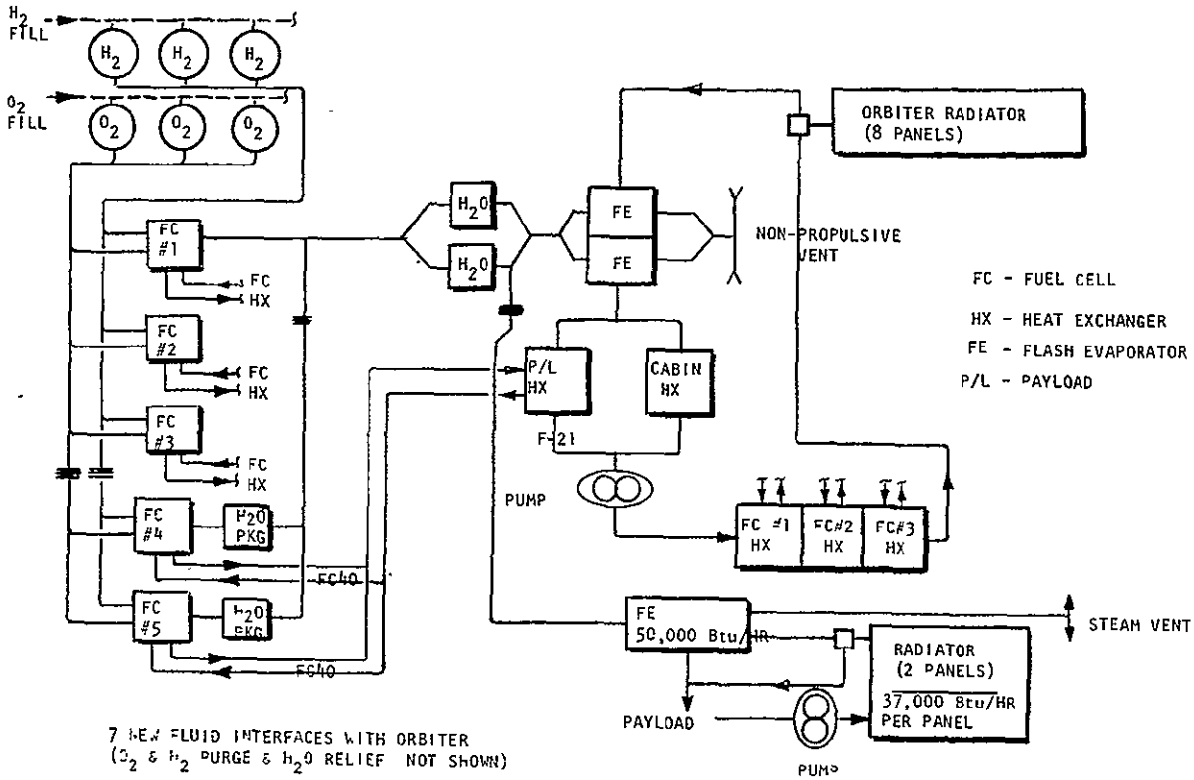
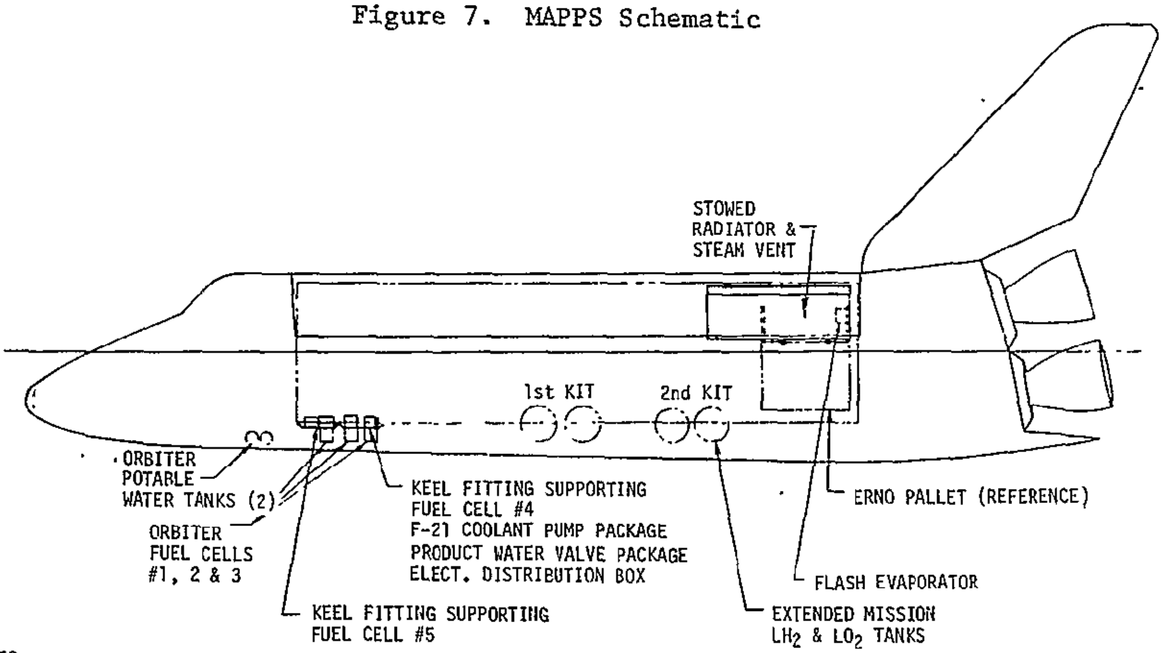


Figure 7. MAPPS Schematic



FEATURES

1 - DESIGNED AS MMSE POWER GENERATION KIT
2 - UTILIZES MAX ORBITER EQUIPMENT
  - FUEL CELLS, RADIATORS, CRYO KITS, WATER TANKS, FLASH EVAPORATOR
3 - NO SPECIAL PALLET REQUIRED
4 - SMALL LOSS OF PAYLOAD BAY VOLUME (~4%)
5 - FLEXIBLE FOR VARIOUS POWER/ENERGY LEVELS
  - 1-5 CRYO TANK KIT
  - 1 or 2 FUEL CELLS
6 - SHORT CRYO LINES
7 - LOW WEIGHT (4327 LB)
8 - LOW PROCUREMENT COST ($3M)

Figure 8. MAPPS Description

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APPENDIX A5

STUDY PLANS FOR FUTURE STUDIES

This Appendix contains study plans for four items which were identified at the mid-term point as requiring substantial study effort in the nature of requirements analysis, conceptual trades and definitions, and programmatics. These are:

1. RTG Cooling Kit
2. Orbiter/Spacelab System Simulator
3. Payload Integrated Pointing System
4. Payload Station Controls and Displays

RTG COOLING KIT DESIGN CONCEPT STUDY

INTRODUCTION

This document defines a plan for conducting a six to seven-month study which will provide the optimum design concept for cooling of RTG's (Radioisotope Thermal Generator) after they have been installed on the spacecraft and installed in the Orbiter payload bay. The kit includes the actual heat rejection/cooling elements and the RTG encapsulator or "muff".

The Problem

Several Shuttle-delivered payloads, including DoD, use RTG's as the means of generating their electrical power. Operation of this type of electrical energy source requires an active cooling system to reject the heat generated continuously. The thermal control system must be capable of satisfying the heat rejection requirements with the payload bay doors both closed (launch to orbit and deorbit) and open (on-orbit, prior to deployment).

There are two basic parts of an RTG cooling kit. The first is the heat dissipation system which previous studies have shown to be a flash evaporator using water as the heat rejection fluid. The second is the encapsulator or muff for the spacecraft RTG unit through which will flow the heat exchange fluid to the cooling unit.

Specific problems are the structural support of the encapsulator assembly, allowances for deflections and vibration, variation in its location, the retraction and stowage mechanism; the location of the bulky heat rejection elements, the routing of the large steam vent line, and the location of the vent outlet considering Orbiter thermal protection system, penetration and payload bay water/ice contamination.

Objective

The objective of the study is to define an optimum RTG cooling kit to satisfy the requirements of all payloads using an RTG. The study will provide information on the design and location, within the Orbiter, of the cooling unit and will also provide the design, method of attachment to the payload, and the concepts for deployment and stowage of the encapsulator. The selected design is to be cost effective and be compatible with the Orbiter restrictions on venting of gases.

SCOPE OF WORK

The study is to be divided into seven major tasks: (1) design requirements and ground rules, (2) encapsulator design, (3) cooling unit design and location, (4) selection of optimum integrated kit configuration, (5) program cost and implementation schedule, (6) conclusions and recommendations, and (7) report preparation.

The following paragraphs present a detailed task-by-task discussion of the technical approach with descriptions of the tasks. The discussion is based on the study flow diagram of Figure 1.

Task 1 - Design Requirements

The purpose of this task is to investigate the NASA and DoD payloads presently planned to use RTG's as a power source, to determine the power level, operating temperature, physical size, location on the payload, and an indication of the structural rigidity of the payload primarily in the area of the RTG. In addition, to identify ground and flight operating timelines with the corresponding heat rejection needs and to indicate the number of payloads (with RTG's) to be flown on each mission and the frequency of the missions.

All ground rules to be used in the conceptual definition of options and in the subsequent design of the selected configuration and the viable alternatives are to be identified. Where the requirements are not available from the payload disciplines, as might be the case for the rigidty of the payload, establish a reasonable ground rule or estimate so that the design effort may proceed. List all limitations of the study. To minimize development costs, Shuttle and other on-going program hardware will be used, where practical.

Task 2 - Encapsulator Design

The problems associated with the design of the encapsulator (muff) are considerable. The method of mounting the unit around the various size and configuration RTG's is complicated by the need to maintain the necessary clearance between the RTG and the muff during all phases of flight. The design is further complicated by the need to remove at least a portion of the muff from the payload and then to stow it in the payload bay for the Orbiter return after the payload has been deployed.

Define the various options with written descriptions and sketches for the RTG encapsulator considering:

a. Amount of shielding
b. Cooling requirements
c. Vibration amplitude of payload
d. Shape and size of RTG
e. Method of mounting to payload or Orbiter structure
f. Method of removal on-orbit
g. Method of attaching to ground and in-flight cooling kit
h. Type of cooling generated in Task 3
i. Stowage during reentry
j. Usage for more than one RTG configuration

Upon completion of the identification of the viable options, list the advantages and disadvantages of each keeping in mind the following criteria:

a. Degree of commonality between payloads
b. Simplicity of on-orbit removal and stowage
c. Weight - unit plus removal mechanism

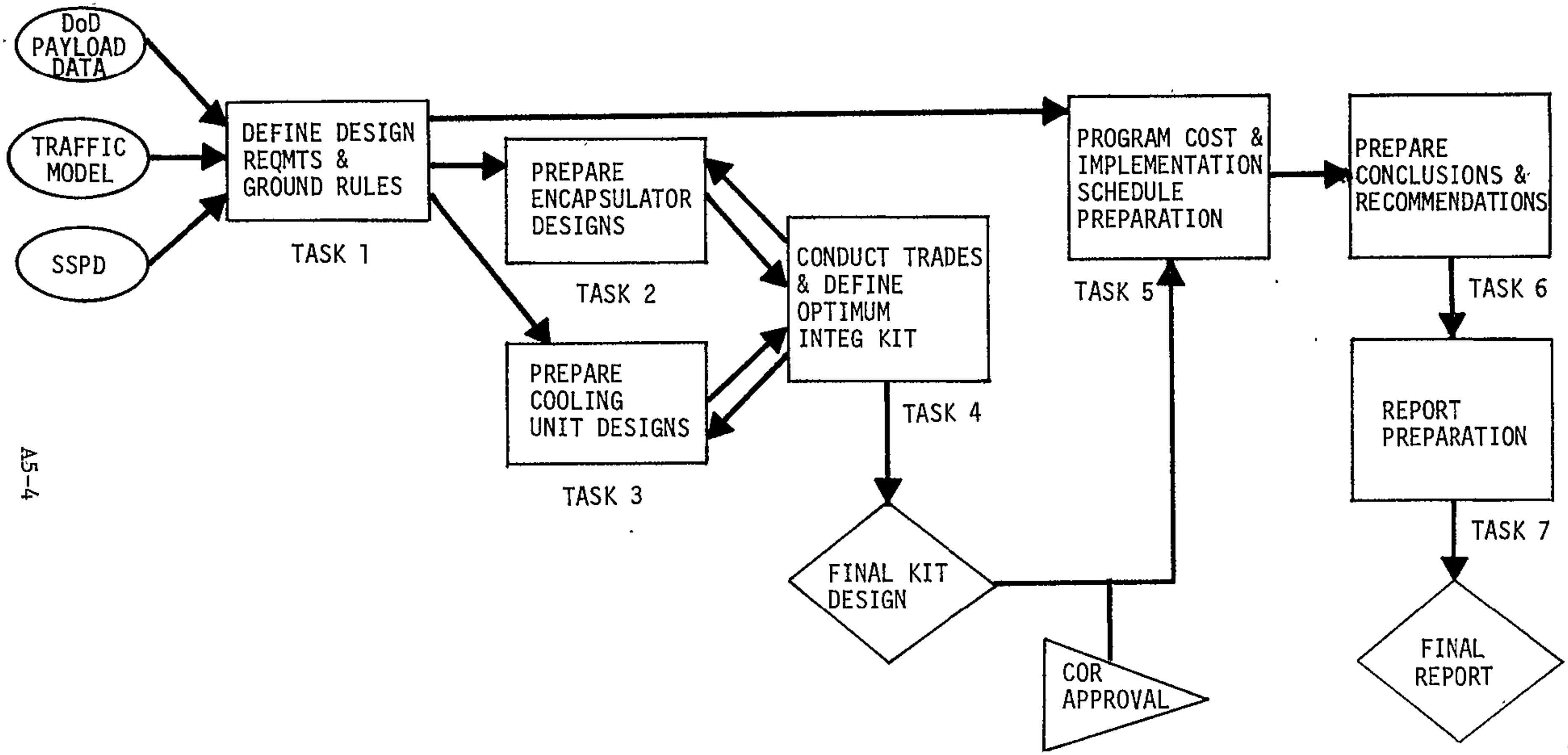

Figure 1. Task Flow Diagram

d. Cost
e. Ease of ground installation and attachment to thermal control system
f. Size

After evaluation of the options, choose one or more as the preliminary configuration for use in the final selection of the integrated system (encapsulator plus cooling elements design and location) in Task 4.

Task 3 - Cooling Unit Design and Location

While the recently completed study of the cooling unit location and configuration (MCR 1405) will be of significant value in this study, re-evaluate the previously selected location to provide a degree of confidence in the compatibility of the cooling unit location with the encapsulator. However, the overriding design requirement for the location of the cooling unit is very probably the location of the penetration of the Orbiter skin with the exhaust steam nozzle of the flash evaporator. The configuration of the unit is, in turn, affected by its location and the design and location of the muff.

Identify and describe (words and sketches) the concepts for the cooling kit or kits to satisfy the requirements of Task 1. One of the major factors in the design of the cooling kit is its location within the Orbiter. Other items affecting the configuration are location are as follows:

a. Heat rejection requirement (flight and ground)
b. Available Orbiter hardware (ground rule will be to use Orbiter hardware wherever practical)
c. Availability of heat rejection capability from the Orbiter

List the advantages and disadvantages of each of the viable options identified and investigated in the previous paragraph. Use the standard criteria of weight, simplicity, reliability, multi-use flexibility, and cost in the process to select one or more of the options to be carried into the next task of selecting the optimum total integrated RTG cooling system.

Task 4 - Selection of Optimum Integrated Kit Configuration

Combine the preliminary encapsulator design(s) from Task 2 and cooling unit concept(s) from Task 3 into an integrated system(s) and evaluate for overall compatibility and ability to satisfy the design requirements and ground rules of Task 1. In the event of a mismatch at this point in the study, re-investigate the less desirable options of Tasks 2 and 3 in order to select the optimum configuration for the total, integrated RTG cooling kit.

Task 5 - Program Cost and Implementation Schedules

Utilizing the requirements of number and frequency of flights, generated in Task 1, determine the number of kits required. Using this number, along with the configuration defined in Task 4, generate first unit, DDT&E, and total program costs.

Prepare an overall program schedule. Include further and more detailed requirements study effort, component development, system development, component and system fabrication, qualification and acceptance testing, and hardware delivery.

Task 6 - Conclusions and Recommendations

Summarize the results of Task 1 (e.g., validity of requirements), Tasks 2, 3, and 4, and those of the overall study. Discuss the feasibility, desirability, and need for a kit or kits. Submit recommendations for the scope, cost, and schedules of the next phase or phases of the program.

Task 7 - Report Preparation

After completion of program costing effort, prepare the final report and briefing. This task is also to cover the monthly progress reports and the mid-term report and briefing effort.

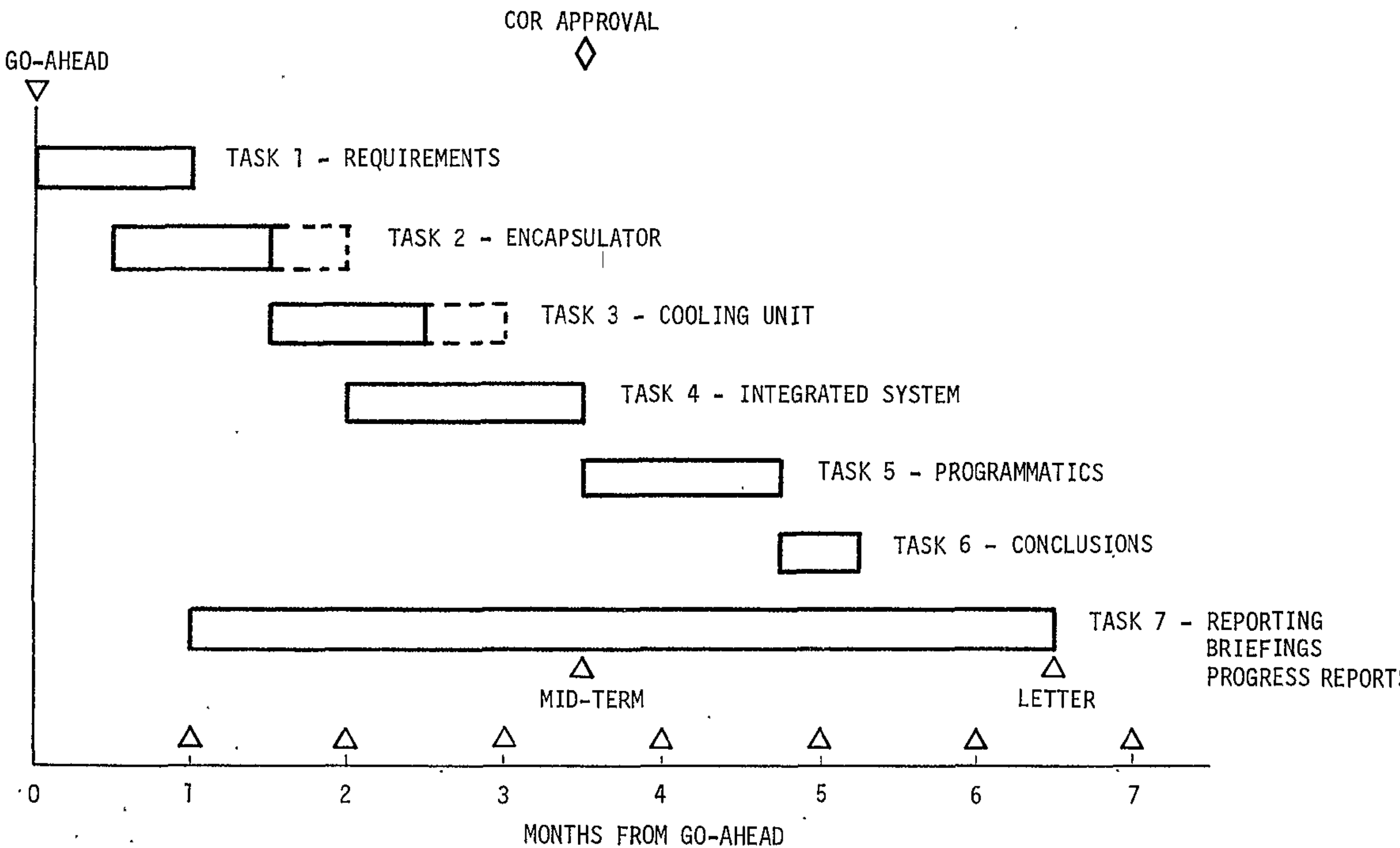

Figure 2. Task Schedules and Milestones

ORBITER/SPACELAB ELECTRONIC SUPPORT SIMULATOR FOR PAYLOAD DEVELOPERS

INTRODUCTION

This document defines a plan for conducting a six to seven-month study that will establish the requirements and select the design concept for the means to provide Orbiter-Spacelab payload interface (input-output-support capability) simulations to aid the development and factory testing of payload elements prior to shipment to integration sites.

The Problem

The general plan is for the Orbiter and Spacelab to provide specified support capabilities for the use of payload elements. Traditionally, it is necessary for the developers of the payload elements to interpret interface specifications then design and build (or procure) test devices in order to simulate inputs, outputs, and support functions. In the case of Orbiter-Spacelab missions, the range of interfaces and support will be fixed (excluding possibilities of occasional redesigns) and repetitiously utilized. This suggests the possibility of standardized test gear that could be utilized repeatedly and save time, cost, and reduce specification mis-interpretation risks as well. Delays and problems at subsequent Orbiter/Spacelab integration could be minimized.

The potential benefits are particularly appealing where the payload element incorporates Orbiter/Spacelab support functions integrally into its functional design. Use of computer support in dynamic control loops is an example. The effects of companion payload elements that draw upon the same support capability must be considered in order to not be adversely impacted and yet avoid over-designs in assuring immunity.

Several categories of interfaces are of concern. These broadly fit into (1) requirements of Orbiter/Spacelab (e.g., C&W and safety command) and (2) payload support needs (e.g., data management, operating displays and controls, up-down link communications, state vector inputs, power and thermal monitoring). The interfaces could be hardwired or data bused, with significantly different impacts upon payload interface hardware design and operation. The relative usage of various types of signals/support depends upon the kind of payload as well as upon Orbiter/Spacelab requirements.

From the above it is seen that the Orbiter and/or Spacelab simulator(s) should provide for a range of interface requirements. Consideration should be made for flexibility of controls and displays and for the use of payload data bus and/or hardwire interfaces. However, it may not be practical to accommodate all possible interface functions AND keep the simulator concept cost effective, easy to use, and easy to handle for the majority of users. Therefore, feasible

options should be defined that meet various capability ranges with evaluations then made for cost effectiveness and ease of use in order to select the best approach.

Objective

The objective of the study is to identify and justify the selection of a means to realistically simulate Orbiter and Spacelab interfaces and support capabilities for individual payload elements. The means will be cost effective, considered as an MMSE item, and be compact and easy to use for typical users. The means shall duplicate a range of input-output-support characteristics that satisfies a large number of payload element classes and covers characteristics such as power, data bus, interconnect cables, controls and displays, and central computer support. The end result is to facilitate payload element development which provides high confidence of integration into Orbiter/Spacelab without problems.

SCOPE OF WORK

Figure 1 summarizes the task logic flow for accomplishing the study. The basic efforts are to define the simulator requirements, develop the potentially feasible design concepts, trade off the design concepts, select the best approach, refine the selected approach, and define any remaining issues to be resolved in order to initiate procurement. The basic tasks are further defined below.

Task 1 - Define Simulator Requirements

The purpose of this task is to identify the types, numbers, and characteristics of interface signals and Orbiter-Spacelab support functions that a simulator may potentially provide. These will be estimated for a range of different types of payload elements or experiments in order to establish the frequency of demand for the numbers and types of interfaces and support. Payload/experiment design data, Shuttle mission traffic models and engineering judgment will be utilized to this end. The categories of signals to be estimated will include those resulting from Orbiter and Spacelab requirements (e.g., C&W monitoring and safety commands) and those that result from the use of support capability from Orbiter and Spacelab (e.g., computer support for control loops, state vector inputs [position, attitude, time, etc.], power, etc.). Characteristics to be determined include such as line/source/load impedances, signal frequencies/voltages, serial digital and discretes characteristics, and computation response/cycle times. The use of hardware and data bus payload-to-Orbiter/Spacelab interfacing will be considered. Orbiter and Spacelab input/output/support characteristics will be separately evaluated with respect to the kinds of payloads they directly support (e.g., some experiments totally interface with Spacelab habitable modules and do not require direct Orbiter support) in order to identify the significant differences and similarities of potential simulated functions. The task results will be displayed in matrices and demand frequency curves as needed to facilitate identification of optimum simulator capability and for making comparisons when establishing simulator concepts/capabilities versus cost-size-flexibility-ease of use.

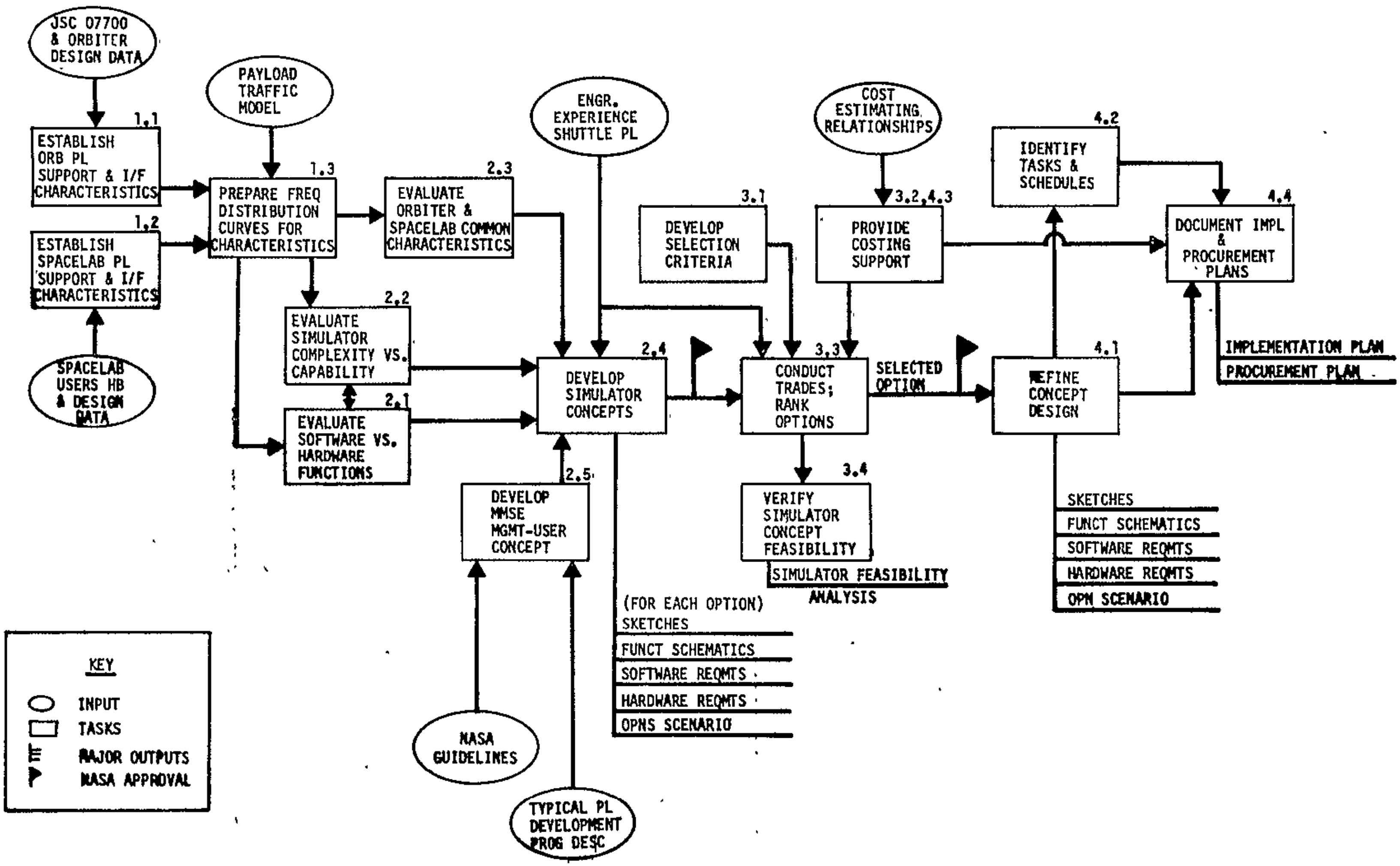

Figure 1. Task Flow Diagram

Task 2 - Identify and Define Simulator Options

This task will utilize characteristics and demand frequency data from Task 1 and potential simulator design and operational concepts will be formulated. Options will be considered that include separate Orbiter and Spacelab simulators and combined Orbiter-Spacelab simulators; the relative feasibilities being determined as a result of the similarity of characteristics for Orbiter and Spacelab interfaces/support functions. Also considered will be simulators that have "add-on" delta capabilities for the less frequently used capabilities as a means to increase range of coverage at lowest cost. The use of self-contained, as well as external computer inputs, in conjunction with necessary input/output channels, will be considered. Simulator programming that serves to verify/debug the operational programming as incorporated into the Orbiter/Spacelab support system programming will also be considered. Each option will be described by sketches, functional schematics, operations logic, and programming logic. Major components as needed to show development and applications costs and complexities will be identified and described. Criteria for managing the use or loaning out of simulators will be defined and the loan periods estimated in order to facilitate estimating inventory requirements for total cost comparisons of options. Options to be evaluated will be subject to NASA approval.

Task 3 - Conduct Trades and Select Option(s)

The options defined in Task 2 will be evaluated and compared in this task in sufficient depth to select the best approach. Overall cost effectiveness and ease of use will be the primary decision criteria to be used in the selection. The cost of not supplying valid capabilities will be included in the trades. Other factors to be considered includes the relative difficulty of implementation, or acquisition, of the required units and the development risks. The results will be presented in a comparison matrix with the required additional rationale to explain the selected approach. Costs will include development, acquisition, and operational/support costs as accured by the MMSE manager and the users who borrow or buy the simulators. Final selections will be subject to NASA approval.

Task 4 - Prepare Procurement Plan

In this task the selected design concept and evaluations will incorporate data from the trade studies of Task 3 as necessary to define the Phase B preliminary design effort. Implementation and procurement plans will be prepared. Any key issues to be resolved during the preliminary design phase will be defined. Programmatic schedules and costs will be refined to permit funding and procurements.

DETAILED TASK BREAKDOWN

Task 1 - Define Simulator Requirements

Objective

1. To understand the types, numbers, characteristics and usage of interface signals/support between payload elements and the Orbiter and Spacelab.

Inputs

1. Orbiter payload support specifications (JSC 07700) and internal design data

2. Spacelab Payload Accommodations Handbook (ESTEC Ref. No. SLP/2104) and data to be requested

3. Payload traffic models

Subtask Breakdown

1. Establish Orbiter payload interface and support characteristics

2. Establish Spacelab payload interface and support characteristics

3. Prepare frequency distributions for characteristics

Expected Results

1. Matrices of signal interface and support characteristics between Spacelab and a range of payload elements and between Orbiter and a range of payload elements

2. The frequencies of occurrence of the characteristics as based upon traffic model flight frequencies for the various types/ classes of payloads

<u>Task 2 - Identify and Define Simulator Options</u>

Objective

1. To develop descriptions of the simulator design concepts and operational usage concepts sufficient to identify significant cost elements, risks, ease of application, and range of applications for trade evaluations.

Inputs

1. Signal characteristics and traffic data (Task 1)

2. Engineering judgement from Shuttle system and MMSE concept experience

Subtask Breakdown

1. Evaluate software versus hardware functions

2. Evaluate complexity versus capability ranges

3. Evaluate Orbiter and Spacelab for common characteristics

4. Develop Orbiter/Spacelab simulator design concepts

5. Develop MMSE management and usage scenarios

Expected Results

1. Sketches, functional schematics, essential design characteristics, operations scenarios, and software requirements descriptions for each simulator concept option. Options will be selected as necessary to resolve (1) the degree of hardware-software application and whether software is (a) part of the simulator or (b) user supplied; (2) the optimum range of capability to be satisfied; (3) whether separate Orbiter and Spacelab simulators or a universal approach is optimum; and (4) whether (a) a basic unit with delta add-ons or (b) more than one simulator configuration or (c) a limited simulation capability is optimum to most cost effectively provide the maximum capability.

<u>Task 3 - Conduct Trades and Select Option(s)</u>

Objective

1. To determine the most cost effective Orbiter-Spacelab simulator approach consistent with the range of payloads needs, ease of use, an MMSE philosophy and risks in development and successful application.

2. To verify that the selected approach is more cost effective and convenient than no MMSE simulator approach.

Inputs

1. Concept option descriptions (Task 2)

2. Cost estimating relationships

3. Shuttle-payload engineering experience

Subtask Breakdown

1. Develop evaluation and selection criteria

2. Provide costing support

3. Conduct trades and rank concepts

4. Verify simulator concept feasibility

Expected Results

1. A trade summary that indicates the selected approach (with rationale)

2. An evaluation of costs, schedules and risks if a simulator concept is not provided as compared to the selected simulator approach

Task 4 - Prepare Procurement Plans

Objective

1. To describe the selected approach and to develop data and planning to a sufficient degree to support preliminary design procurement.

Inputs

1. Selected options and related data (Task 3)

Subtask Breakdown

1. Define design concept
2. Identify implementation tasks and schedules
3. Provide costing support
4. Document implementation plan and Phase B procurement requirements

Expected Results

1. A detailed description of the selected design concept
2. An implementation plan document with task definitions, schedules, and cost estimates
3. A Phase B procurement requirements document

STUDY SCHEDULE

Figure 2 depicts the period of performance for tasks and subtasks. Task 1 utilizes data already available to summarize germane Orbiter and Spacelab characteristics. The Spacelab effort takes longer since its design is less firm. The data is summarized for easy use in Task 2. Cognizance effort to update significant Orbiter and Spacelab characteristics data will be maintained until final documentation starts.

Task 2 can begin to examine basic issues immediately since the essential characteristics of Orbiter-Spacelab interfaces and support cabilities are known. However, conclusions are not finalized until the latest data from Task 1 is available. Development of specific simulator concepts depends upon Task 1 data and results of 2.1, 2.2, 2.3, and 2.5. Subtask 2.5 specifies the operational system in which the simulator will be utilized (as important to overall cost and effectiveness factors).

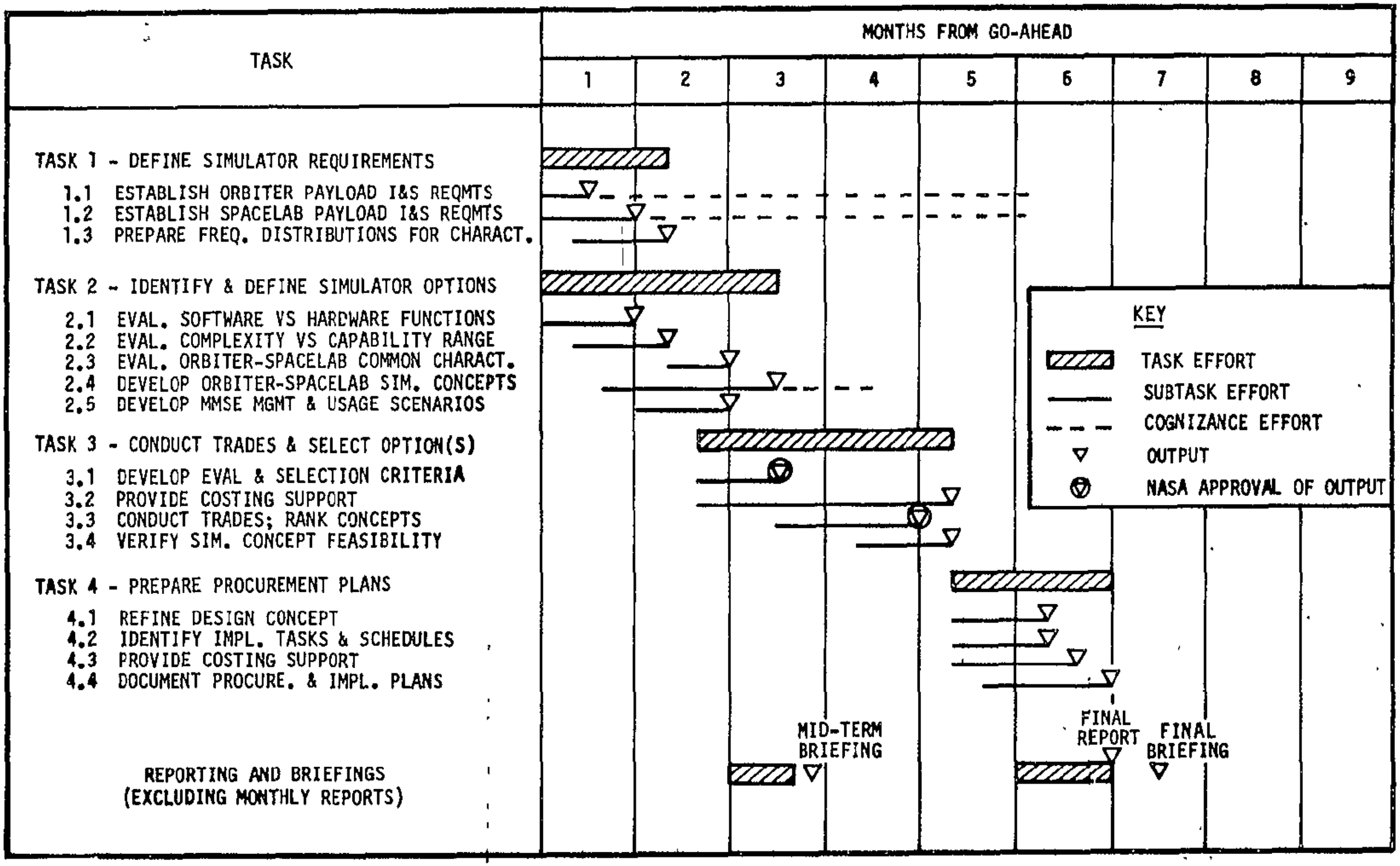

Figure 2. Task Schedules and Milestones

The mid-term briefing to NASA will describe the options to be subjected to detailed trades. The options are subject to NASA approval and some effort in Subtask 2.4 to refine options due to NASA inputs may occur.

The trade studies in Subtask 3.2 will be based on criteria formalized in Subtask 3.1. The ranking and concept selection in 3.3 will be coordinated with NASA. Subtask 3.4 will independently establish the cost and effectivness elements for users who must develop their own simulation capabilities. Then the results will be compared to the case of the simulator selected in Subtask 3.3. Subtask 3.2 provides specialist costing support, utilizing available estimating relationship data and methods.

The subtasks of Task 4 can largely be done in parallel. Final clean up of cost data is completed after the design has been refined and procurement/ implementation tasks have been defined. The final report, primarily editing the outputs of the other subtasks outputs, can begin relatively early for the technical writeup sections. Cost and schedule data is incorporated during the last week before publication. An additional period of time for NASA review before final publication could be provided but is not shown in Figure 2. A final briefing for NASA and the projected users will complete the study effort.

PAYLOAD INTEGRATED POINTING SYSTEM COMPONENTS STUDY

INTRODUCTION

This document defines a plan for conducting a nine-months study that will establish a list and define the requirements for MMSE items that can be used to mechanize a variety of payload pointing control systems.

The Problem

Generalized MMSE components that may be usable in pointing control systems were identified in preceding studies (NAS8-30847 out of MSFC and NAS9-14598 out of JSC). However, closed loop pointing mechanizations have not been evaluated to establish definitive component requirements and to verify the suitability of potentially usable items identified to date. The NAS8-30847 study identified a microcomputer and a high performance inertial measuring unit (IMU) as well as star, sun and earth trackers/sensors. The NAS9-14598 STS-MMSE study concluded that the Orbiter Star Tracker (OST) could satisfy a large range of payload pointing; possibly satisfy almost all stellar pointing needs with appropriate modifications. Use of the STS OST or modified OST (MOST) in place of the NAS8-30847 non-STS star tracker recommendation appears to reduce the IMU and computer requirements significantly. The IMU and computer requirements are affected by the modifications to OST. It is not, however, clear which OST modifications (if any) are optimum considering the variations possible in impact upon the IMU, computer, and other components for the range of payload requirements to be supported.

In addition, it appears that a star tracker may be useful for some earth and sun pointing missions given appropriate operational or design installation considerations. A detailed evaluation of these considerations is needed to determine if it is feasible/desirable to use star trackers for such missions, thereby possibly reducing the number of different types of required sensors.

Component technology advances also must be considered. Computer support supplied by Orbiter or Spacelab has been generally assumed. However, low cost microcomputers (MOS, LSI, magnetic bubble memories, etc.) are becoming available which are smaller, lighter, and consume much less power than Orbiter vintage computers. Their use promotes payload independence of development and operations, which may be important since the ability of the central computers to meet requirements of high performance control loops, considering data transfer delays and possible variations in response due to other payload demands, is not firmly established. Therefore, the use of independent microcomputers versus central computers needs evaluation in order to fully establish the MMSE potential for microcomputers. More than one kind of microcomputer for pointing and other control applications may be applicable.

The feasibility of using a modified Orbiter IMU was not considered in the NAS9-14598 STS-MMSE study since the requirements for an MMSE IMU identified by

the NAS8-30847 study required an order-of-magnitude better performance than the Orbiter IMU capability. However, detailed pointing studies may indicate a use for the Orbiter IMU as MMSE. More recent state-of-the-art inertial components such as laser or electrostatic suspended type gyros should also be considered.

Another consideration is the use of large control moment gyros (CMG's) to fine-control Orbiter attitude. This would increase the potential number of flights that could use strap-down pointing instead of gimbal systems and could minimize Orbiter RCS pollution of payload sensors.

A variation to direct CMG control of Orbiter is the use of a semi-isolated platform (such as by springs) which is controlled by CMG's. The latter has been studied in the past but should be reviewed to establish application in light of the latest payload information. The number of independent platform control mechanizations could be reduced on missions that have multiple pointed instruments. This would impact the type and number of MMSE required.

It is evident that the frequent need for payload pointing and the large number of variables to be considered for the mechanizations requires an in-depth integrated pointing study to define MMSE items.

Objective

The objective is to identify a minimum list of standard component items that can most effectively (cost, convenience, and time) meet a wide range of payload pointing accuracies, stabilities, and stability rates. The items will be compatible with Orbiter strap-down pointing and standard large and small platforms to the degree practical. Consideration of the users problems will be given full attention. Modifications to existing items (such as the Orbiter Star Tracker) will be defined where necessary. STS-MMSE will be considered wherever applicable.

SCOPE OF WORK

This section of the study plan presents a detailed task-by-task discussion of the technical approach. The Figure 1 task flow diagram depicts the task interrelationships in the basic approach.

Task 1 - Identify Payload Pointing Requirements

In this task, the latest payload pointing requirements will be assembled and evaluated for frequency of occurrence as a function of pointing characteristics. Potential combinations of pointed and non-pointed payload elements on the same mission will be stablished and the combined limits and requirements described. The latest payload description documents and payload studies for optical, IR and UV astronomy, advanced technology laboratory, high energy astrophysics, and solar physics will be used to define and understand the pointing control needs and operational constraints. Data describing pointing systems that have been studied to handle various payloads will be acquired and reviewed. Any design requirements applicable to specific payloads will be identified and confirmed through responsible payload offices.

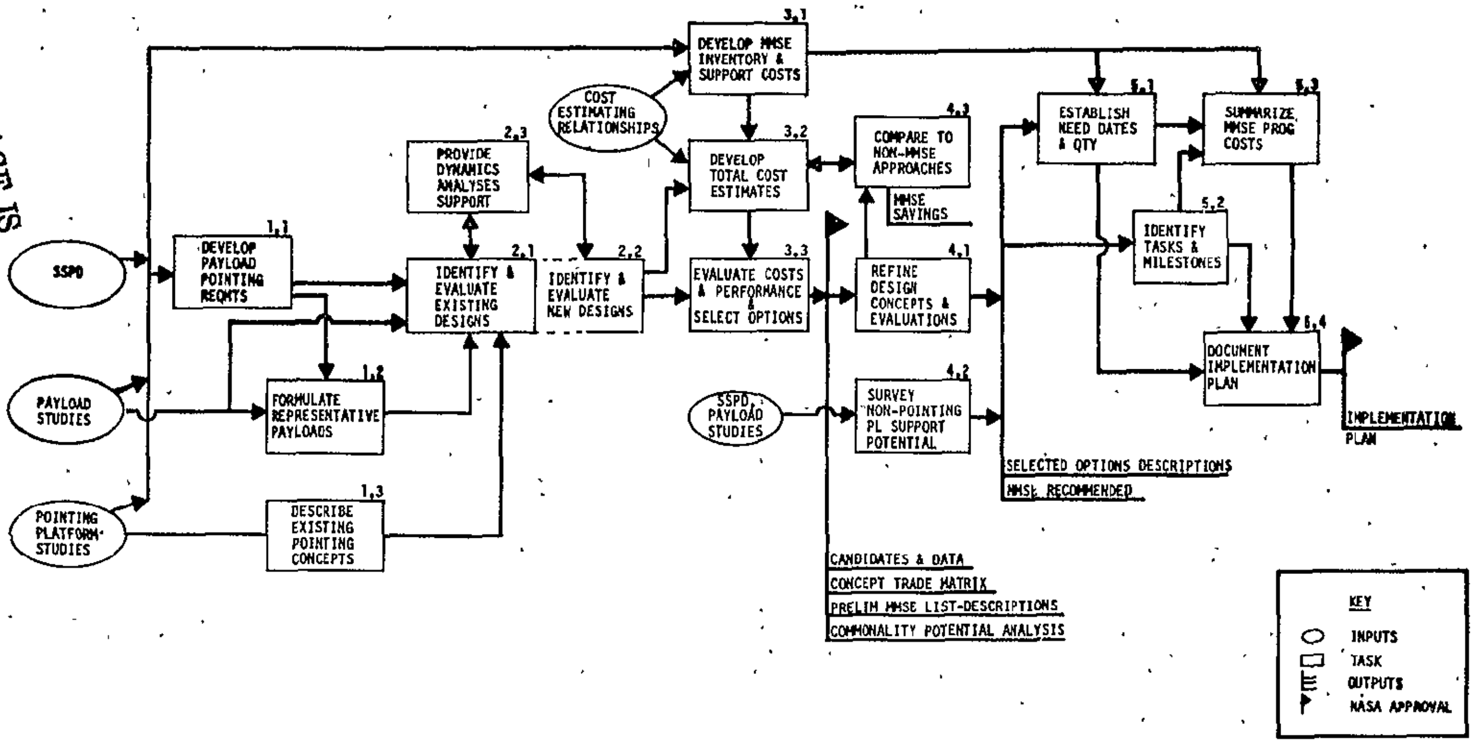

Figure 1. Study Flow Diagram

Task 2 - Develop Pointing Concepts and Designs

Existing Orbiter strap-down and isolated platform concepts will be identified and all necessary components and interfaces that are needed to complete the control mechanizations will be defined. The capability of one or more of the concepts to meet the requirements and operational conditions developed in Task 1 will be evaluated. Additional concepts/variations/component applications will be identified where classes of requirements may potentially be met by more cost effective means. The use of large CMG's to control Orbiter attitude and/or semi-isolated large platforms are examples of additional concepts. The capability of each design to meet performance requirements will be verified by simplified dynamic control system analyses, which will be used to select/verify component parts. Various design studies (e.g., Ball Brothers SIPS, Rockwell alternate astronomy pointing system analyses, ERNO IOG large gimbal platform) as available, will be used as inputs. Payload agencies will be contacted to ascertain the firmness of commitment to various concepts they have/are studying. The primary objective is to identify potential MMSE concepts as needed to completely mechanize around the platforms (including Orbiter) to be utilized.

Task 3 - Compare Candidates and Select Optimum Approaches

Compare the candidate control concepts and specific designs developed in Task 2 for costs and effectiveness in covering the range of payload pointing requirements and for ease of integrating into Orbiter/Spacelab payloads. The goal is to identify a minimum number of MMSE items that are needed to mechanize the greatest number of payload pointing schemes, consistent with cost effectiveness and user convenience. The reasons why users would use MMSE in lieu of their potential independent selections will be identified.

The performance, total cost estimates (development, acquisition and operation), and design and operational integration requirements will be developed for each option and suboption. Reliability and maintenance/support will be included in operational costing. A MMSE management scenario will be developed and the user requirements for applying MMSE in design, debugging, performance verification, and integration into Orbiter payloads will be considered. The number of MMSE needed in inventory and their costs will then be estimated.

From comparisons of cost and effectiveness factors for the options and suboptions evaluated, a baseline concept design will be selected for each class of pointing requirements.

Task 4 - Finalize Designs and Evaluations

This task will incorporate any changes necessary to optimize each needed pointing control design concept as selected from Task 3. The dynamic analysis and component characteristics requirements will be refined and updated to verify selections. Any modifications required (e.g., to OST) on existing components will be fully defined. The MMSE inventory estimates, management scenarios, and cost factors will be refined. The cost deltas to perform equivalent support to payloads without an MMSE concept will be determined for comparisons. Other, non-pointing functions that could be handled by proposed MMSE items (e.g., microcomputer/magnetic bubble data storage devices which could perform other control/data management functions, store star catalog data, etc.) will be investigated.

Task 5 - Develop Implementation Plan

For the designs from Task 4, an MMSE implementation plan will be prepared. The key tasks to be completed and the time frames for performance to meet initial flight dates will be developed. Operational inventory requirements will be determined. Design, development, User design integration, and payload integration tasks will be estimated. The dollar appropriations required from the start of design (if any) to implementation and operation, including support facilities costs, will be developed.

DETAILED TASK BREAKDOWN

Task 1 - Identify Payload Pointing Requirements

Objective

1. To develop a detailed, updated understanding of all classes of pointed payload element requirements.

2. To establish the possibilities, limits, and restrictions in combining pointed payload elements with other pointed and non-pointed payload elements.

3. To develop an understanding of current payload agency plans/thoughts on meeting their/other requirements.

Inputs

1. 1975 (later if available) SSPD

2. ATL, astronomy, solar, high energy physics pointing studies

3. Personal contacts with payload offices TBD

4. Pointing platform studies (SIPS, EOG, TBD)

Subtask Breakdown

1. Develop (acquire, review, extract and confirm) pointing requirements data

2. Formulate representative Orbiter payloads (containing pointed elements)

3. Describe existing platform concepts-designs

Expected Results

1. Matrix of pointing requirements by payload name, class, and flight frequency

2. Range of integrated Orbiter payload element combinations and associated characteristics as a function of pointing/operational ground rules (Orbiter strap down; platform with small [+5°-10°] freedom; platforms with large [$\sim\pm90°$] freedoms).

3. Descriptions of applicable pointing platforms that are/were studied and the committed and uncommitted elements in such approaches (e.g., a platform design concept may be firm but the control loop electronics may be defined only in terms of requirements).

Task 2 - Develop Pointing Concepts and Designs

Objective

1. To understand the scope of potentially usable pointing control concepts applicable to various requirement classes and admissable mission operational ground rules.

2. To develop a list with characteristic requirements of possible MMSE candidates which have commonality potential.

3. To describe performance capabilities and sensitivities of candidate concepts and design variations.

Inputs

1. Identifiable payload pointing studies (SIPS, IOG, Rockwell studies, isolated platform study, others TBD)

2. Payload requirements and data from Task 1

3. Orbiter pointing control characteristics

4. Component (sensors, trackers, IMU's, electronics, computers) data sheets on physical, performance, and cost characteristics (commercial, military, STS)

Subtask Breakdown

1. Identify and evaluate existing designs

2. Identify and evaluate new designs

3. Provide dynamic performance analysis support

Expected Results

1. Description and performance evaluations of each viable pointing concept and design variation. Interface description and requirements.

2. List of components used and applicable characteristics, including modifications required on existing designs. Identification of Orbiter/Spacelab support required.

3. Potential commonality of components among pointing system design.

Task 3 - Compare Candidates and Select Approaches

Objective

1. To select the pointing concepts and associated MMSE components as needed to perform a broad range of payload pointing, considering the necessary fixed elements such as platform design concepts.

Inputs

1. Dynamic analyses for concepts evaluated in Task 2

2. Component characteristics and cost data

3. User development, test, and integration cycle programmatics.

Subtask Breakdown

1. Develop MMSE inventory and support estimates

2. Develop total cost estimates (each option and suboption)

3. Compare costs and performances and select options

Expected Results

1. Scenario descriptions for typical management and usage of pointing control MMSE

2. Total cost developments and deltas for all Task 2 options

3. Matrix of operations and selection criteris with the selections identified.

4. Matrix of MMSE components as a function of selected pointing approaches.

Task 4 - Finalize Designs and Evaluations

Objectives

1. To refine the designs and evaluations of the selected basic pointing concepts and MMSE and compare to non-MMSE approach costs.

2. To identify other than pointing-payload support functions that could be performed by pointing control MMSE.

Inputs

1. Task 3 definition data - selected approaches
2. Additional components information
3. General payload support requirements data (from which to identify potential uses of MMSE in addition to pointing control)

Subtask Breakdown

1. Additional design and evaluation (of selected approaches and MMSE)
2. Survey additional payload support function potential
3. Compare MMSE and non-MMSE approaches

Expected Results

1. Final recommended pointing options and MMSE with the supporting rationale
2. Potential applications of the listed MMSE for other support functions (e.g., uses for microcomputers)
3. Component usage estimates for non-MMSE approaches (for comparison purposes)
4. Estimated cost savings and other advantages of the MMSE approach over the non-MMSE approach

Task 5 - Prepare Implementation Plan

Objective

1. To develop and describe the needed tasks, schedules and costs that will result in the required inventory of pointing control MMSE, associated support system, and MMSE management capability.

Inputs

1. Data from prior tasks of this study
2. Experience with similar developments

Subtask Breakdown

1. Establish need dates and quantities
2. Identify tasks and milestones

3. Summarize MMSE program costs

4. Document implementation plan

Expected Results

1. An integrated development plan that identifies all additional tasks that lead to an operating MMSE program for the recommended items.

STUDY SCHEDULE

Figure 2 shows the study task time phasing and key milestones. The first two months will be primarily devoted to acquiring and reviewing germane payload studies and description documents and contacting payload agencies by telephone and/or visits to assure understanding of their intents, limitations, and flexibilities for approaches they are studying. Also, MMSE management-user concepts will be discussed with payload agencies. Results will be summarized for subsequent use in the study. Cognizance of potential Users plans will continue until just after the mid-term review to assure current data for the final study report.

Task 2 will begin prior to completion of Task 1 as soon as firm Users design concepts information is verified. New concepts will be developed if needed to "fill holes" in the ability of currently planned designs to provide a broad spectrum of support. New concepts that could accomplish broader requirements more cost effectively will also be considered. Specialized control systems analysts will support dynamic analyses and the selection of control loop components. Cognizance of User studies will continue until after the mid-term in order to consider any new approach suggestions.

Task 3 is inherent to a degree in Task 2 and preliminary efforts will be performed a month or so before the main effort starts. Subtask 3.1 independently develops a logical plan for the usage of MMSE and its maintenance and can begin early as convenient. Completion of subtask 3.2 must occur a week or two before the total cost estimates can be completed. Concept/design comparisons and selections for MMSE will be completed after the mid-term briefing to allow for possible NASA inputs and approval of the selections.

Task 4 is a relatively short effort to make refinements as necessary to the selected approaches and to evaluate the benefits of an MMSE approach as opposed to a non-MMSE approach. A closer look at optional system components and the definition of modifications to such as STS OST and IMU are other prime objectives of Task 4.

The implementation plan for the recommended MMSE will be completed in Task 5, using data from study Tasks 1 through 4. The tasks and key milestones of subsequent efforts as necessary to obtain a working MMSE inventory of pointing control components will be developed in Task 5.

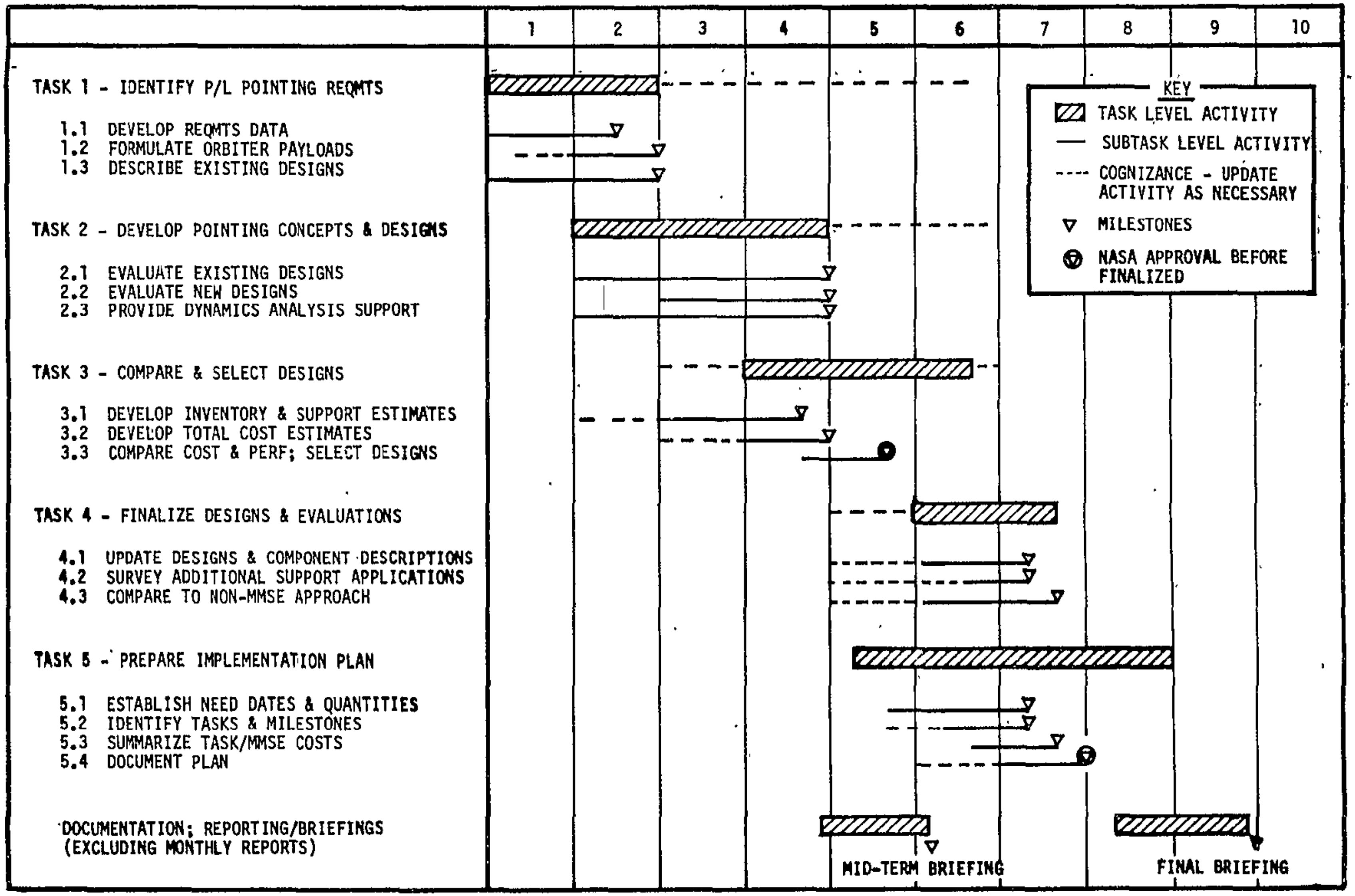

Figure 2. Study Schedules and Milestones

PAYLOAD STATION (PS) CONCEPT STUDY

INTRODUCTION

This document presents a plan for conducting an eight to nine-month study to define the detailed physical and functional interfaces between payloads and the PS and to develop a PS design concept which can be used to procure the necessary design effort.

The Problem

Orbiter payloads (in the bay and when free-flying following deployment or prior to recovery) require interfacing with the PS in order to facilitate payload control and monitoring. The specific control and monitoring requirements vary with the payload, although many essential characteristics appear to be similar at least on a class basis. Classes of payloads include Spacelab habitable modules, Spacelab pallets, automated (free flyer) spacecraft, and S/C-IUS/kick stage combinations. On-orbit maintenance operations may also impose unique requirements. In general, control and monitoring from the Orbiter aft crew station and from ground stations must be accommodated and data outputs from primary sensors must be managed. Functions to be performed include initialization, test and checkout, state vector updates, and payload health monitoring. C&W and safety commands may be supplemented. In the case of all-pallet payloads, control and operation of specific experiments will also be accomplished from the PS. Interaction with operation of the payload bay doors, C&W, the RMS, Orbiter maneuvers, and other payloads must be considered and integrated.

It is desirable to maximize flexibility and minimize interface complexities for users. However, in order to reduce costs, development risks, lead times, and Orbiter integration impacts, it appears desirable to standardize the control and display functions as much as practical. At the same time, the volume and panel space available in the aft crew stations are limited. The human factors arrangement of panels is important. Operational coordination and monitoring between the PS and the on-orbit station, mission specialist station (MS), and caution and warning (C&W) panels must be considered.

It also appears desirable to consider a payload-dedicated microprocessor/computer within the PS in order to provide payload support and to mechanize the control and display functions. Payload support functions could include RMS operation for one and/or two arms. Additionally, the military indicates a desire to fly mixed payloads with other users in order to reduce launch costs. A separate computer may be needed for classified payloads. The ability to handle such mixed payloads must, therefore, be considered.

Finally, studies indicate that hardwire payload bay cabling may result in design installation and electrical problems that could be alleviated by a data bus system. Therefore, potential operation with such systems should be considered.

In summary, the problem is that of arriving at an optimum PS functional and physical design approach that recognizes the users needs and desires while minimizing the amount of unique PS hardware and software needed for specific flights. In addition, the various uncertainties and potential interface requirements and support needs must be accommodated.

Objective

The objective of the study is to fully define the requirements and limitations on the PS, define and evaluate PS design approach options, and select and further design the overall best approach. A Phase B preliminary design procurement concept specification and a program plan, with backup rationale, are specific objectives.

SCOPE OF WORK

Figure 1 provides a task flow diagram that illustrates the task flow logic of the study. Details are provided below.

Task 1 - Establish Operations and Functional Requirements

The purpose of this task is to establish the range of PS functional capabilities needed to satisfy the range of Orbiter payload requirements, to establish the relative traffic demand rates for the significant PS capability requirements, and to establish the interface and support constraints imposed by Orbiter design, Orbiter requirements, and potential companion payloads. The combined requirements will be determined for any feasible mix and number of the various classes of payloads (i.e., Spacelab modules, pallets, automated spacecraft, IUS-S/C, kickstage-S/C, maintenance modules, etc., from NASA, military, commercial, and other agencies). The degree of commonality and the potential for sharing of PS capabilities by payloads will be determined in recognition of limited PS panel space and total volume and the desirability to minimize the amount of PS component changeout for each flight. Requirements for unique capability will also be identified in order to determine the necessary provisions for PS flexibility for accommodating all potential payloads. The results and data from prior human factors PS panel layout studies by JSC, as well as other applicable control and display studies for NASA, military, and commercial payload accommodations, will be incorporated into the capability requirements evaluations. The latest Orbiter design data will be used to establish physical and functional interface criteria and support capabilities. Emphasis will be placed on coordinating with representative users of the PS to ensure acceptance and compatibility.

Task 2 - Synthesize PS Design Options and Suboptions

Using the requirements evaluations from Task 1, synthesize the various significantly different potential approaches for meeting the full range of payload PS support requirements. Basic approaches considered will range from maximum capability and universality to "core" capabilities that have provisions for adding "delta" capabilities. Also considered will be the use of Orbiter and/or separate PS - contained computers/processors. The special needs, if any, to accommodate the shared flights by the different agencies (NASA, DoD,

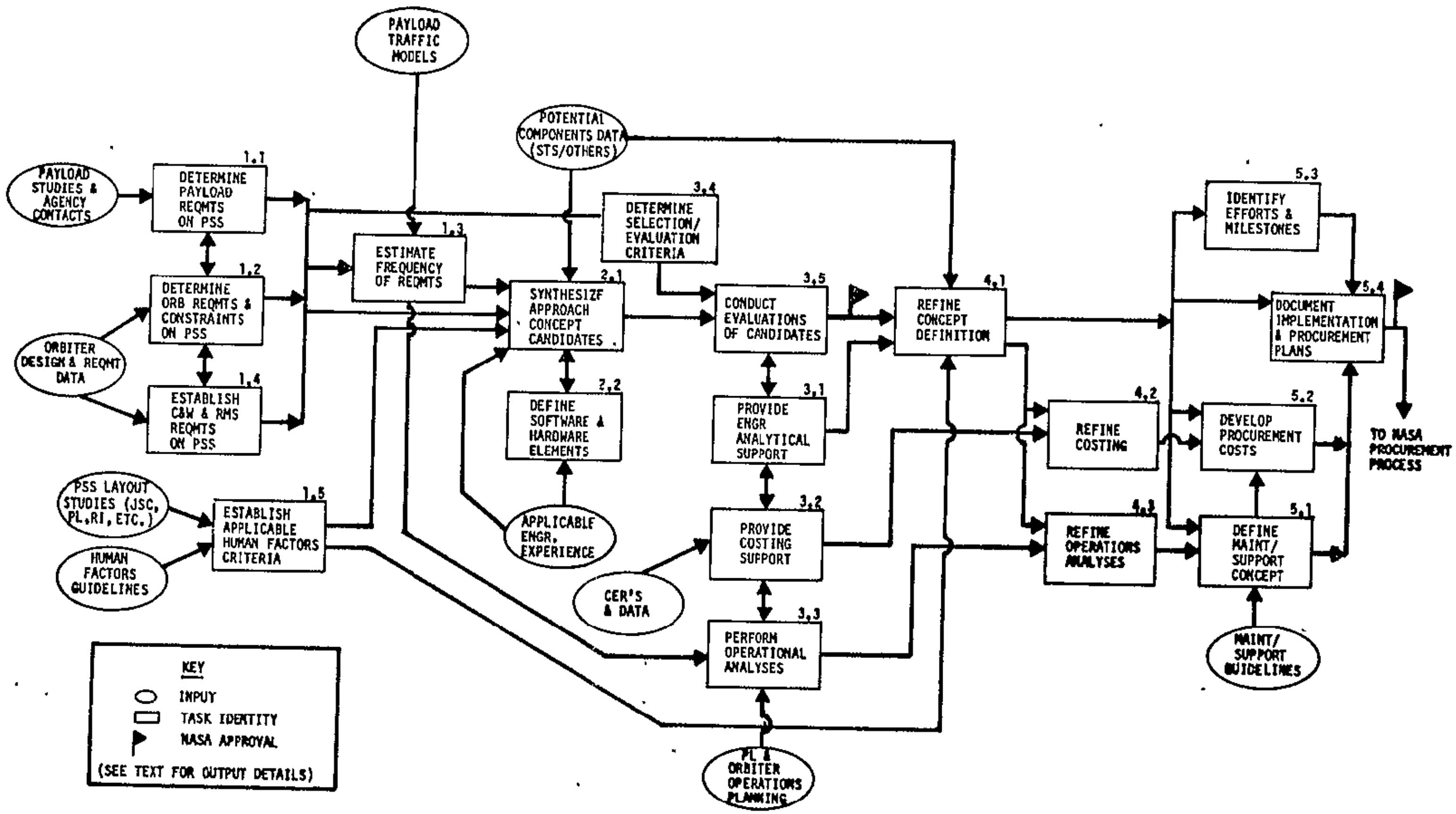

Figure 1. Task Flow Diagram

commercial, other) and different project offices will be considered. The effects of using multiplexed interfaces with payloads (except mandatory hardwiring) versus hardwiring will be considered. The goal will be to maximize the suitability and ease of integration by the payloads/users and overall cost effectiveness to the degree practical while observing all necessary constraints. The capability to provide other aft crew station payload related support such as Remote Maneuvering System (RMS) and end effector (e.g., spacecraft spin mechanisms) control and C&W processing will be addressed. The generation of concepts will consider the availability of suitable STS (and other) components as identified in this task (e.g., microcomputers, displays, display electronic units, etc.). At least three significantly different options will be developed for Task 3 tradeoff analyses. These will be selected by comparisons and engineering judgment. The concepts will be developed and described to the level of detail needed to facilitate total cost evaluations, programmatic analyses, and user application evaluations. Concept descriptions will include system and component layouts, functional logic diagrams, and the essential characteristics of suitable off-the-shelf and/or new components. Delta descriptions of viable suboptions will also be provided.

Task 3 - Evaluate Candidates and Select Approach

This task will define evaluation criteria and define and evaluate each candidate provided by Task 2 in sufficient detail to identify the best approach. Evaluations will confirm technical feasibility, providing any necessary mechanization and sizing analyses to verify the capability of components to handle functions, rates, and ranges. Software and integration/checkout steps will be identified for all operational phases for the user/integrator/operator. Reliability comparisons between options will be made. All significant total cost elements will be estimated, utilizing available cost data and/or cost estimating relationships. The percentage of the mission model traffic accommodated by each hardware/software configuration and that traffic expected to require payload-unique accommodations will be determined for each candidate. The results of the comparison analyses will be displayed in a matrix format and the recommended approach, which strikes the best balance between cost, ease of use, and mission traffic commonality, will be indicated.

Task 4 - Refine Design and Operational Concepts (of Selected Approach)

This task will continue the design definition on the selected approach from Task 3 as required to develop a preliminary design procurement specification and to refine hardware cost estimates. Features of the discarded candidates from Task 3 and other data may be incorporated to achieve the most optimum approach. Component definitions and design analysis will be extended in order to select specific preliminary design approaches and/or to define the detailed design trade studies to be performed in the Phase B effort. The result will be a list of major hardware components, layouts, and PS design requirements. A detailed scenario of how the PS users interface with and utilize the PS from payload design through operation, considering potential mixed payloads, will also be developed and documented.

Task 5 - Prepare Implementation and Procurement Plans

This task places emphasis on documenting the main products of the study, a procurement specification and a program implementation plan. Cost and design requirements from Task 4 will be augmented with a programmatics evaluation to establish all implementation program efforts, key milestones, and total budget allocations needed to support the first and subsequent Orbiter flights. The number of PS items to be procured and the procurement phasing to meet operational, integration, payload development, and spares needs will be determined. The maintenance/support system concept will be defined and the related costs and schedules will be separately identified.

DETAILED TASK BREAKDOWN

Task 1 - Establish Operations and Functional Requirements

Objective

1. To identify all constraints and desired/required capabilities that should be considered in the design approach of the PS and to indicate the flight frequency at which each capability is expected to be required.

Inputs

1. Contacts with payload design agencies (NASA, military, commercial) and the NASA PS working group

2. Payload traffic model

3. Payload concept/design/programmatic studies as accessible

4. Current Orbiter payload interface and support design criteria, including physical layouts, RMS control requirements and C&W requirements

5. JSC 09343, "Space Shuttle Payload Accommodations on the Aft Deck", January 1975

6. JSC 09321, "Orbiter 102 Feb. 1975 PDR Payload Interfaces Team Documentation"

7. Human Factors Design Guidelines (TBD)

Subtask Breakdown

1. Determine potential payload requirements for the PS

2. Determine Orbiter requirements and constraints on the PS

3. Estimate commonality and frequency of requirements

4. Establish C&W and RMS interface/support

5. Establish Control and Displays human factors criteria.

Expected Results

1. Matrix of requirements versus payloads/payload classes. Also includes estimate of flight frequency for each requirement.

2. Defintion of payload classes and representative payloads

3. Human factors guidelines for controls and displays layouts and functional operation

Task 2 - Synthesize PS Design Options and Suboptions

Objective

1. To identify all significantly different functional and design approaches that could potentially prove to be the most effective in meeting mixed payload PS needs.

Inputs

1. Requirements, constraints, and flight frequency data from Task 1

2. STS and other applicable components capability data

3. Applicable engineering judgement, ingenuity and experience

Subtask Breakdown

1. Synthesize system concepts

2. Define hardware and software elements

Expected Results

1. Three or more concept descriptions (sketches, functional diagrams, component lists)

2. Descriptions of suboptions to the basic concepts

3. Gross comparative traffic accommodation and feasibility analyses to justify the subsequent detailed Task 3 trade evaluations

Task 3 - Evaluate Options and Select Approach

Objective

1. To select the most effective basic approach in terms of overall costs, ease of use, and support capabilities.

Inputs

1. Descriptions of options from Task 2
2. Payload traffic, requirements, and desires data from Task 1
3. Constraints and human factors data from Task 1 if required
4. Cost data and cost estimating relationships (CER's)

Subtask Breakdown

1. Provide engineering analytical support
2. Provide costing support
3. Perform operational analyses (support systems, reliability, human factors, traffic accommodations)
4. Determine evaluation criteria
5. Conduct evaluation of options

Expected Results

1. Trade matrix comparing options on the basis of cost, feasibility, risk, reliability, ease of operation, ease of integration, ease of support, and traffic accommodated.
2. Recommended approach with additional rationale as warranted.

Task 4 - Refine Design and Operational Concepts

Objective

1. To develop the selected design concept to greater detail as needed to verify its selection and to define requirements for a formal preliminary design program.

Inputs

1. Applicable data for the selected approach from Tasks 2 and 3
2. Characteristic data for applicable components

3. Costing data and CER's

4. Human factors guidelines from Task 1

Subtask Breakdown

1. Refine concept definitions

2. Refine costing

3. Refine operational analyses

Expected Outputs

1. Final concept description that identifies components, provides system and PS conceptual layouts, and describes functions and complete requirements that specifies preliminary design procurement.

2. Verification analyses that support the selection.

Task 5 - Prepare Implementation and Procurement Plans

Objective

1. To perform a maintenance/support system and programmatic analysis to establish key implementation efforts and schedules, estimate procurement costs, and document the implementation plan and design procurement (Phase B) specification.

Inputs

1. Refined design and operational evaluation data from Task 4 for the selected approach

2. Maintenance/support concept guidelines

3. Payload mission model

Subtask Breakdown

1. Define maintenance and support requirements

2. Develop procurement costs

3. Identify efforts and milestones

4. Document implementation plan and procurement specification

Expected Results

1. Implementation plan containing task descriptions for all efforts and procurements needed to bring the PS system into the operational inventory and maintain it. Schedules, milestones, and estimated costs for each effort by year will be provided to facilitate procurement budget planning.

2. Procurement specification that describes the PS concept to be developed and its interfaces and functional operation. Detailed design trade issues will be defined. Specifies all firm requirements relative to concept, functions, and design.

STUDY SCHEDULE

Figure 2 shows the period of performance for each task and subtask and indicates the milestones for internal and external outputs. Recommended NASA approvals for the selected PS approach and the procurement data are also shown.

Task 1 is key in that a firm understanding with a representation of users on PS requirements, from the payload point of view, is felt necessary. Coordination with the NASA PS working group and individual agencies is envisioned to obtain payload desires and assure that they are realistic with respect to Orbiter constraints. Orbiter constraints are somewhat in flux and continuing interaction with Orbiter in-house design groups is planned. Cognizance of payload and Orbiter activities bearing on PS requirements will continue until study wrap-up begins so that any significant changes can be factored into design approach considerations. The issues of defining support by the PS for C&W and RMS operation will be resolved and human factors criteria will be developed in order to state all requirements of the PS function. The several studies performed by JSC and others regarding control and display layouts will be utilized for reference.

Task 2 provides payload and Orbiter engineering experience that can define potentially feasible options to meet requirements developed in Task 1. Since software may be as important an issue as hardware, separate subtasks were set up to emphasize the two aspects of concept synthesis and provide for necessary experienced personnel. Task 2 can overlap Task 1 to a considerable degree based upon current knowledge of requirements.

Subtask 3.5 is the focus of Task 3 efforts. Specialized support is obtained from subtasks 3.1, 3.2, and 3.3 in order to evaluate system design trade issues, define operational scenarios, and estimate relative costs of options. Subtask 3.4 provides the basis of the candidate selection in Subtask 3.5 and will be coordinated with the COR. Except for Subtask 3.4, Task 3 is a continuation of Task 2.

A mid-term briefing to NASA is planned at the end of Task 3 in order to explain the options considered and to provide rationale for the recommended option. Cognizance efforts will be continued after the briefing in order to include any NASA comments. NASA concurrence on the recommended option is planned.

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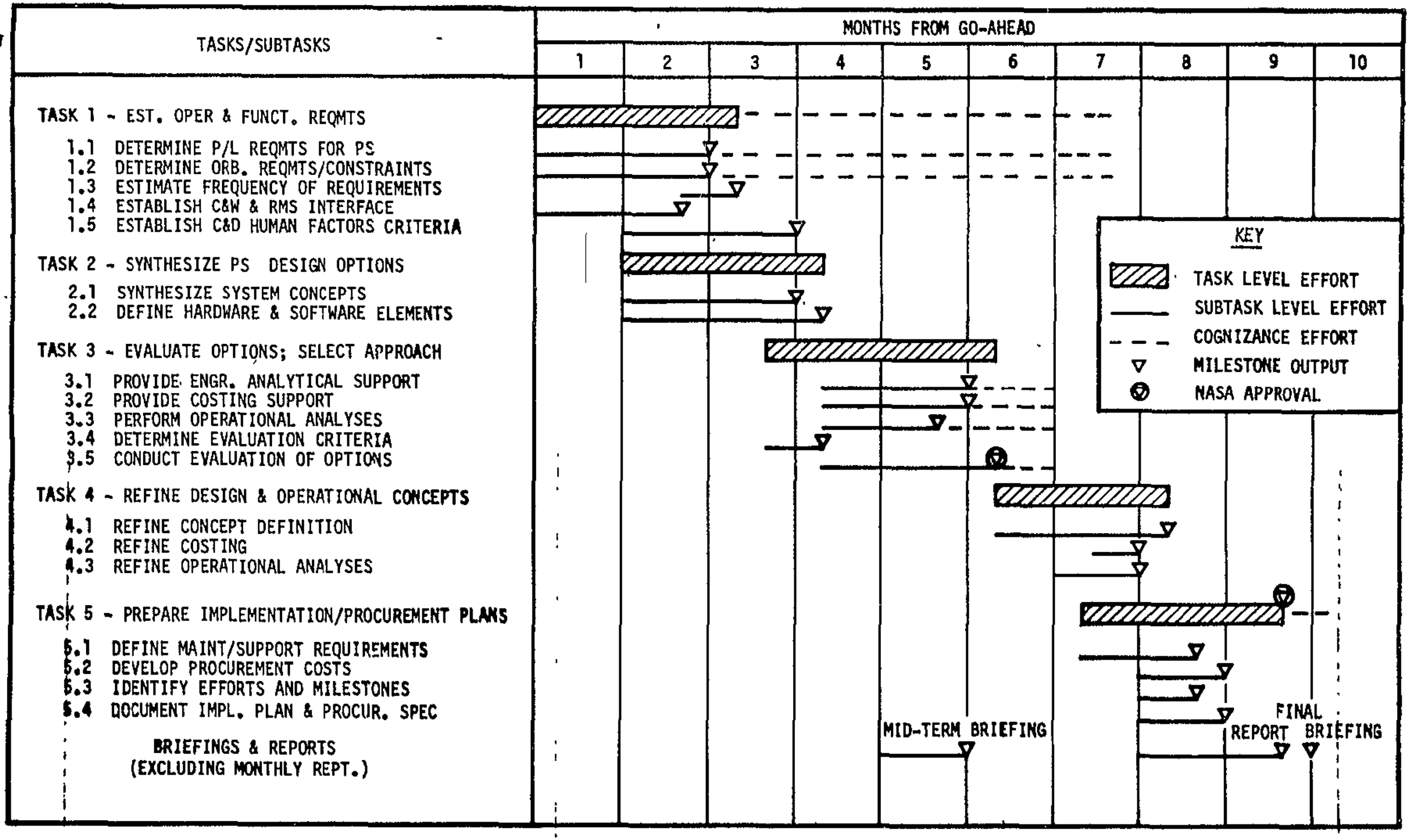

Figure 2. Task Schedules and Milestones

The Task 4 effort to refine the selected design is a continuation of Task 3. Subtasks draw upon specialized support personnel to carry out a more detailed and absolute approach than the comparative, higher level evaluations of Task 3.

Task 5 can overlap Task 4 to initate the final programmatic analysis subtasks and to begin documentation of the implementation plan and procurement specification. These primary study outputs are subject to NASA approval. The final report documentation effort is considered to be a part of Task 5. The final briefing is planned for about the end of the ninth month after study go-ahead.